CHAPTER 2. PART 29
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

SUBPART B - FLIGHT

GENERAL

AC 29.21. § 29.21 (Amendment 29-24) PROOF OF COMPLIANCE.

a. Explanation.

(1) This section provides a degree of latitude for the FAA/AUTHORITY test team in selecting the combination of tests or inspections required to demonstrate compliance with the regulations. Compliance must be shown for each combination of gross weight, center of gravity, altitude, temperature, airspeed, rotor RPM, etc. Engineering tests are designed to investigate the overall capabilities and characteristics of the rotorcraft throughout its operational envelope. Testing will identify operating limitations, normal and emergency procedures, and performance information to be included in the FAA/AUTHORITY-approved portion of the flight manual. The testing must also provide a means of verifying that the rotorcraft's actual performance, structural design parameters, propulsion components, and systems operations are consistent with all certification requirements.

(2) Section 21.35 requires, in part, that the applicant show compliance with the applicable certification requirements, including flight test, prior to official FAA/AUTHORITY Type Inspection Authorization (TIA) testing. Compliance in most cases requires systematic flight testing by the applicant. After the applicant has submitted sufficient data to the FAA/AUTHORITY showing that compliance has been met, the FAA/AUTHORITY will conduct any inspections, flight, or ground tests required to verify the applicant's test results. FAA/AUTHORITY compliance may be partially determined from tests conducted by the applicant if the configuration (conformity) of the rotorcraft can be verified. Compliance may be based on the applicant's engineering data, and a spot check or validation through FAA/AUTHORITY flight tests. The FAA/AUTHORITY testing should obtain validation at critical combinations of proposed flight variables if compliance cannot be inferred using engineering judgment from the combinations investigated.

(3) Performance tests include minimum operating speed (hover), takeoff and landing, climb, glide, height-velocity, and power available. Certain other performance tests, such as Category A, are conducted to meet specific requirements. Detailed performance test procedures and allowable extrapolation or simulation limits are contained in the respective paragraphs in this order.

(i) Hover tests are conducted to determine various combinations of altitude, temperature, and gross weight for both in-ground-effect (IGE) and, if required, out-of-ground effect (OGE) conditions. From these data the hover ceiling may be calculated.

(ii) Takeoff and landing tests are conducted to determine the total distance to takeoff and land at various combinations of altitude, temperature, and gross weight.

(iii) Climb tests establish the variations of rate-of-climb at the best rate-of-climb or published climb airspeed(s) at various combinations of altitude, temperature, and gross weight.

(iv) Height-velocity tests are conducted to determine the boundaries of the height versus airspeed envelope within which a safe landing can be accomplished following an engine failure.

(v) Power available tests are conducted to verify or reestablish the calculated installed specification engine performance model on which published performance is based.

(4) The purpose of rotorcraft stability and control tests is to verify that the rotorcraft possesses the minimum qualitative and quantitative flying qualities and handling characteristics required by the applicable regulations. In order to assess the handling qualities, standardized test procedures must be utilized and the results analyzed by accepted methods. Section 29.21(a) allows calculation and inference which includes extrapolation and simulation, whereas § 29.21(b) requires demonstration of controllability, stability, and trim. Combinations of §§ 29.21(a) and 29.21(b) may be used to show compliance to the operating envelope limits. Test methods and equipment are described in individual paragraphs of this advisory circular.

b. Procedures.

(1) Efforts should begin early in the certification program to provide advice and assistance to the applicant to insure coverage of all certification requirements. The applicant should develop a comprehensive test plan which includes the required instrumentation.

(2) The tests and findings specified in paragraph a(3) above are required of the applicant to show basic airworthiness and probable compliance with the minimum requirements specified in the applicable regulations. After these basic findings have been submitted and reviewed, a Type Inspection Authorization, or equivalent, can be issued. The FAA/AUTHORITY will develop a systematic plan to spotcheck and confirm that compliance with the regulations has been shown. The test plan will consider combinations of weight, center of gravity, RPM and cover the range of altitude and temperature for which certification is requested.

AC 29.21A. <u>§ 29.21(Amendment 29-39) PROOF OF COMPLIANCE</u>.

a. <u>Explanation</u>. Amendment 39 added § 29.83 which changes the requirements for determination of landing distance for Category B rotorcraft. This amendment requires landing distance to be determined with all engines operating within approved limits.

b. <u>Procedures</u>. The guidance material presented in paragraph AC 29.21 continues to apply.

AC 29.25. <u>§ 29.25 (Amendment 29-12) WEIGHT LIMITS</u>.

a. <u>Explanation</u>.

(1) This section is definitive and specifies criteria for establishing maximum and minimum certificating weights. These weights may be based on those selected by the applicant, design requirements, or the limits for which compliance with all applicable flight requirements has been shown.

(2) Typical requirements that may establish the maximum and minimum weight limits include:

Maximum: Structural limits, performance requirements, stability, and controllability requirements.

Minimum: Autorotative rotor RPM, stability, and controllability requirements.

(3) <u>Jettisonable External Cargo</u>.

(i) Paragraph (c) was added by Amendment 29-12 to provide, in the certification standards, a basis for approving an increase in gross weight (exceed standard limits) that would be an external jettisonable load. The attachment device standards were moved from Part 133 (Amendment 133-5) to Parts 27 and 29. Section 29.865, “External load attaching means,“ now contains the standards, including design features, for the attaching devices. Cargo hoists and hooks were envisioned. Prior to these amendments, type design approvals were made under Part 133 and the policy in Review Cases Nos. 37 and 55 of FAA Order 8110.6 whenever the standard limits were exceeded.

(ii) In the preamble of Amendment 29-12 (Proposal 2-99, 41 FR 55454, December 20, 1976) the agency stated, in part, that “...§ 29.25(c) is intended to provide only a total weight standard for approving the rotorcraft structure (and propulsion systems) for operation under Part 133.” As indicated in § 29.865, fatigue substantiation of the external cargo attaching means is not required. The rotorcraft structure, rotors, transmissions, engines, etc., are subject to evaluation under § 29.571 for external cargo approval whenever the “standard” structural limitations are exceeded (Review Case Nos. 37 and 55).

(iii) Whether or not the standard limitations are exceeded, the flight characteristics evaluations/standards of § 133.41 are appropriate even for engineering approval. This Part 133 standard is also applicable for the individual operator to obtain his operating certificate. The operator may use an FAA/AUTHORITY approved RFM supplement for external load operations to prepare a rotorcraft load combination flight manual required by § 133.47.

b. Procedures.

(1) It may not be possible to demonstrate quantitatively all the flight requirements at the minimum weight because of test instrumentation requirements. The test team must be assured that the rotorcraft complies with the applicable requirements at the lowest permissible flying weight. This evaluation may be done qualitatively, with the test instrumentation removed, and with minimum crewmembers if no critical areas exist or are anticipated. Additionally, reasonable extrapolation may be warranted. However, if critical areas at minimum flying weights are apparent, extrapolation should not be permitted.

(2) Whenever a gross weight increase (§ 29.25(c)) is requested, a TIA evaluation is necessary to evaluate the new limitations and ensure that § 133.41 for typical or representative cargo shapes and weights (density) is satisfactory. All possible combinations of weights and shapes are not evaluated. The representative configurations may be noted in the RFM or RFM supplement for the operator’s information. Sections 133.41 and 133.47 must be satisfied by the individual operator for the particular case at hand. The approved RFM or RFM supplement should provide the necessary limitations and any other information about the representative cargo configurations evaluated. Section 133.41 also permits the operator to obtain approval of additional and unique cargo configurations provided the approved limitations are observed. Paragraph AC 29.1581 concerns the RFM and its contents.

(3) See paragraph AC 29.571, § 29.571, for fatigue substantiation and external cargo considerations.

(4) Refer to AC 133-1A, Rotorcraft External-Load Operations in Accordance with FAR Part 133, October 16, 1979, for further information on airworthiness and flight manual policy.

AC 29.27. § 29.27 (Amendment 29-3) CENTER OF GRAVITY LIMITS.

a. Explanation.

(1) This regulation is definitive and requires that the center of gravity limits be defined. Proof of compliance with all applicable flight requirements is required within

the range of established CG's. Along with the longitudinal CG limits, the lateral CG limits should either be established or determined to be not critical.

(2) Ballast is usually carried during the flight test program to investigate the approved gross weight/center of gravity limits. Lead is the most commonly used form of ballast during rotorcraft flight testing although other types of ballast, such as water, may serve just as well. Water may have the added benefit of being jettisonable during critical flight test conditions. Care must be taken regarding the location of ballast. The strength of the supporting structures should be adequate to support such ballast during the flight loads that may be imposed during a particular test and for the ultimate inertia forces of § 29.561(b)(3). Of critical importance is the method of securing the ballast to the desired locations. To avoid any undesired in-flight movements of the ballast, a positive method of constraint is mandatory. The flight test crews should also visually verify the amount, location, and integrity of the ballast. The effects of mass moment of inertia on the flight characteristics due to the ballast locations should also be considered. The mass moment of inertia of the test rotorcraft should, to the extent possible, be the same as that expected in normal, approved loadings, especially during tests involving dynamic inputs.

b. Procedures.

(1) Center of gravity locations and limits are of prime importance to rotorcraft stability and safety of flight. The primary concern is establishment of the longitudinal center of gravity limits. Lateral center of gravity limits with respect to longitudinal center of gravity limits are also important. The design of the rotorcraft is usually such that approximate lateral symmetry exists. This lateral symmetry can be upset by lateral loadings resulting in the necessity to establish lateral center of gravity limits. There are two characteristics which may be seriously affected by loading outside the established center of gravity limits; these are stability and control. The established center of gravity limits must be such that as fuel is consumed, it is possible for the rotorcraft to remain within the established limits by acceptable loading and/or operating instructions.

(2) Structural limits may restrict the maximum forward longitudinal center of gravity limits. However, in most cases it is the maximum value established wherein adequate low speed control power exists to meet such requirements as § 29.143(c). Likewise, the maximum aft center of gravity limit may be a "structural limit," but it usually is determined during flight test after the rotorcraft's handling qualities tests have been conducted. Additional items which may influence the maximum aft center of gravity limits may be malfunctions of automatic stabilization equipment, excessive rotorcraft attitudes during critical phases of flight, or adequate control power to compensate for an engine failure.

(3) Lateral center of gravity limits have become more critical because of the ever increasing utilization of the rotorcraft for such things as unusual and unsymmetric lateral loads, both internal and external. Maximum allowable lateral center of gravity limits have also influenced the results of the unusable fuel determination.

(4) Summarizing, it is of prime importance that longitudinal and lateral center of gravity limits be determined so that unsafe conditions do not exist within the approved altitude, airspeed, ambient temperature, gross weight, and rotor RPM ranges. All relevant malfunctions must be considered.

AC 29.29. § 29.29 (Amendment 29-15) EMPTY WEIGHT AND CORRESPONDING CENTER OF GRAVITY.

a. Explanation. The empty weight of the rotorcraft consists of the airframe, engines, and all items of operating equipment that have fixed locations and are permanently installed in the rotorcraft. It includes fixed ballast, unusable fuel, and full operating fluids except water intended for injection in the engines.

(1) Fixed ballast refers to ballast that is made a permanent part of the rotorcraft as a means of controlling the certificated empty weight CG.

(2) Compliance with paragraph (b) of § 29.29 is accomplished by the use of an equipment list which defines the installed equipment at the time of weighing and the weight moment arm of the equipment.

b. Procedures.

(1) Determination of the empty weight and corresponding center of gravity is primarily the responsibility of a manufacturing inspector and is normally made on a production rotorcraft rather than a prototype. If a manufacturer wishes to avoid weighing each production rotorcraft and has been issued a production certificate, the manufacturer may make a detailed proposal defining the procedures used to establish an empty weight and CG When the proposal is approved, the manufacturer will weigh the first five to ten production rotorcraft and show that the rotorcraft will be within ±1 percent on empty weight and ±0.2 inches on CG After this procedure is established, the empty weight and CG may be computed except that at regular intervals a rotorcraft will be weighed to ensure the tolerances are still being maintained; e.g., one in ten rotorcraft.

(2) For prototype and modified rotorcraft, it is only necessary to establish a known basic weight and CG position (by weighing) from which the extremes of weight and CG travel required by the test program may be calculated. See AC 91-23A, Pilots Weight and Balance Handbook, June 9, 1977, for a sample weight and balance procedure.

(3) The weight and balance should be recalculated if a modification (or series of modifications) to the rotorcraft results in a significant change to the empty weight. Additionally, this change in empty weight should be reflected with the weight and balance information contained in the Rotorcraft Flight Manual (RFM) or Rotorcraft Flight Manual Supplement (RFMS).

c. Ballast Loading and Type.

(1) Ballast loading of the rotorcraft can be accomplished in any manner to achieve a specific CG location. It is acceptable for such ballast to be mounted outside the physical confines of the rotorcraft if the flight test objectives are not affected by this arrangement. In flight test work, loading problems will occasionally be encountered in which it will be difficult to obtain the desired CG limits. Such cases may require loading in engine compartments or other places not designed for load carrying. When this condition is necessary, care should be taken to ensure that local structural stresses are not exceeded or that the rotorcraft flight characteristics are not changed due to increased moments of inertia by attaching the ballast to extreme CG locations which may not be designed for the added weight.

(2) Two types of ballast that may be used in loading are solids or liquid. The solids are usually high density materials such as lead while the liquid usually used is water. In critical tests, the ballast may be loaded in a manner so that disposal in flight can be accomplished. In any case, the load should be securely attached in its loaded position so shifting or interference with safety of flight will not result.

AC 29.31. § 29.31 REMOVABLE BALLAST.

a. Explanation. This regulation provides the option of using removable ballast for operational flights to obtain center of gravity locations that are in compliance with the flight requirement of this Part. Fixed ballast used for flight operations after type certification must be documented in the type design data. Removable ballast is used primarily on small rotorcraft to control the CG with different passenger loadings although this regulation does permit its use on transport rotorcraft. If removable ballast is used, the rotorcraft flight manual must include instructions regarding its use and limitations. See paragraph AC 29.873 for information on ballast provisions.

b. Procedures. None.

AC 29.33. § 29.33 (Amendment 29-15) MAIN ROTOR SPEED AND PITCH LIMITS.

a. Explanation.

(1) General. This rule requires the establishment of power-on and power-off main rotor speed limits and the requirements for low rotor speed warning.

(2) Power-On. The power-on limits should be sufficient to maintain the rotor speed within these limits during any appropriate maneuver expected to be encountered in normal operations throughout the flight envelope for which certification is requested. A power-on range of approximately 3 percent has in the past been the minimum range required due to engine governor and engine operating characteristics. With the introduction of advanced engines and electronic engine controls, there may not be a

need for a range, but one fixed value may suffice. Transient power-on values may also be acceptable provided they are substantiated.

(3) Power-Off. The power-off rotor speed limits should be sufficient to encompass the rotor speeds encountered during normal autorotative maneuvers except for final landing phase (touchdown) for which rotor RPM may be lower than the minimum transient limit for flight, provided stress limits are not exceeded. The limits should also be sufficient to cover the ranges of airspeed, weight, and altitudes for which certification is requested. It is not the intent of the rule to require the minimum and maximum limit values in conjunction with extremes such as maximum/minimum weights and/or high altitude. The minimum and maximum rotor speed requirements should be thoroughly evaluated at normal operation environment; i.e., at altitudes between approximately sea level and 10,000 feet, temperatures not at extremes, and weights as necessary for other tests and as required to readily establish the limit rotor speeds. Spot checks of the autorotative requirements should be made at the extremes of the flight envelope and environmental conditions during normal tests at those conditions. Under conditions where high autorotative rotor speeds may be encountered, it is acceptable for the pilot to adjust the controls to prevent overspeeding of the rotor. At light weight combined with low altitudes and extreme cold temperatures, the normal low pitch setting may not be sufficient to maintain autorotational rotor speed values within limits. If this occurs, the manufacturer may elect to adjust the low pitch stops as a maintenance procedure at extreme ambient conditions provided the flight and maintenance manuals clearly present the rigging requirements and procedures. There must be sufficient “overlap” of ambient conditions between configurations such that rerigging is not required whenever ambient temperature and surface elevation change slightly. Any down rigging of the low pitch stop must continue to ensure adequate clearance between controls and other rotorcraft structure and should be evaluated during flight test. Both the power-on and power-off limits may also be established by encountering critical flapping limits in some approved flight conditions such as high airspeed or sideward flight.

(4) Additional RPM Ranges. Some applicants have elected to certify their aircraft with additional RPM ranges in an attempt to realize additional performance during certain flight conditions or maneuvers such as Category A OEI continued and rejected takeoffs and balked landings. Such additional RPM ranges have been found acceptable as long as all pertinent FAR requirements are fully substantiated for operation in that range. The substantiation should include drive system endurance and flight test verification of performance and flight characteristics during applicable maneuvers, in the additional RPM range. The FAA/AUTHORITY does not define additional RPM ranges as transient since all applicable requirements must be satisfied for approval of that range.

(5) Low Speed Warning. If it is possible under expected operating conditions for the rotor speed to fall below the minimum approved values, the requirement exists for a low rotor speed warning. This warning is required on all single-engine rotorcraft and on multiengine rotorcraft where there is not an automatic increase in remaining

engine(s) power output upon failure of an engine. Although today's multiengine rotorcraft do not require a low rotor speed warning according to the rule, essentially all have warning systems installed. If the minimum power-on and power-off rotor speed limits are different, the warning signal should be at the higher speed, normally the power-on minimum rotor speed. One rotorcraft has a warning system cutout if the collective is full down, and others have other warnings on the engine speed to indicate engine failure. All of these related warning systems must be evaluated with emphasis on ensuring adequate rotor speed.

b. Determination and Testing. Refer to paragraph AC 29.1509 (§ 29.1509) for additional information on rotor limits determination and testing.

SUBPART B - FLIGHT

PERFORMANCE

AC 29.45. § 29.45 (Amendment 29-24) PERFORMANCE - GENERAL.

a. Explanation.

(1) Amendment 29-39 adopts new and revised airworthiness standards for the performance of transport category rotorcraft. As part of this change, several sections within the performance section of Subpart B were renumbered. The performance section of this guidance material has been organized for easy use with rotorcraft certified before or after this amendment. To achieve this, some of the guidance material has been duplicated under different paragraph numbers. A statement at the beginning of each of these paragraphs indicates where other pertinent information can be found.

(2) Section 29.45 lists some of the rules and standards under which the performance requirements are to be met. This paragraph will provide general guidelines that may be used throughout a flight test program. It is impossible to find ideal test conditions and there are many variables which affect the flight test results that must be taken into account. Some of these variables are wind, temperature, altitude, humidity, rotorcraft weight, power, rotor RPM, center of gravity, etc. The test results should be analyzed and expanded by approved methodology within the guidelines of this paragraph. A thorough knowledge of the testing procedures and data reduction methods is essential and good engineering judgment must be used to determine applicable test conditions.

(3) Performance should be based on approved engine power as determined in paragraph b(4) below and not on any transient limits. Approved transient limits are basically for inadvertent overshoots of approved operational limits. Any sustained operation in these transient limit areas usually require some form of special maintenance. However, for such demonstrations as rejected and continued OEI Category A takeoffs and height-velocity (HV) determination, low rotor speeds have been authorized based upon additional structural and drive train substantiation (see paragraph AC 29.33).

(4) Where variations in the parameter on which a tolerance is allowed will have an appreciable effect on the test, the results should be corrected to the standard value of the parameter; otherwise, no correction is necessary.

b. Procedures.

(1) Winds For Testing.

(i) Allowable wind conditions will vary with the type of test and will also be different for different types and gross weight rotorcraft. For example, higher winds can usually be tolerated for takeoff and landing distance tests than for hover performance. Likewise higher winds can sometimes be tolerated during hover performance testing on large, heavy rotorcraft with high rotor downwash velocities than for smaller rotorcraft with rotor downwash velocities. Generally, unless the effects of wind on hover performance tests can be determined and/or accounted for, hover performance testing should be conducted in winds of 3 knots or less.

(ii) Past experience has shown that a steady wind of 0 to 10 knots will result in acceptable takeoff and landing performance if distances are corrected for the winds measured during these tests. This is not the case for vertical takeoffs and landings. To obtain consistent and repeatable vertical performance data, the same general wind criteria used to obtain hover performance; i.e., up to 3 knots, should be adhered to for vertical performance determination. In actuality, a rotorcraft may exhibit reduced IGE hover performance in winds from 3 to 15 knots due to partial immersion of the main rotor in its own vortex. Since the height-speed envelope determination is affected by wind just as vertical takeoff and landing performance are, the same allowable winds for testing should be adhered to for HV testing; i.e., 0 to 3 knots.

(iii) As can be seen from the foregoing, there is no such thing as an exact allowable wind for a particular test or rotorcraft. The flight test team must decide on the allowable wind for each condition based on all available information and their engineering judgment. The following summary of allowable wind conditions are given for general guidance only:

(A) Hover performance - 0 to 3 knots.
(B) Conventional takeoff and landing - 0 to 10 (data to be corrected)
(C) Vertical takeoff and landing - 0 to 3 knots
(D) Height-velocity - 0 to 3 knots

(iv) A means should be provided to measure the wind velocity, direction, and ambient air temperature at the rotor height for any particular tests. The wind effects on required runway length for takeoff and landing distances may be shown in the flight manual.

(v) Full wind credit may be given for conventional takeoff and landing field lengths. This credit should not be more than the nominal wind component along the takeoff or landing path opposite to the direction of flight.

(2) Altitude Effects. Using FAA/AUTHORITY-approved methodology, hover, takeoff, and landing, performance may be extrapolated and/or interpolated from test data up to a maximum of ±4,000 feet. Experience has shown that IGE handling qualities, height-velocity, and engine operating characteristics should not be extrapolated more than approximately 2,000 feet density altitude from the test altitude.

Cruise stability/controllability tests should be evaluated at least at two different altitudes, the lowest practical altitude and approximately the highest cruise altitude requested for approval. This can allow an interpolation of approximately 10,000 feet. As in all testing, extrapolation and/or interpolation should only be considered if all available information and engineering judgment indicate that regulatory compliance can be met at the untested conditions.

(3) Altitude Limitations.

(i) Explanation.

(A) Two altitudes are normally presented in the RFM to define the operating envelope of a rotorcraft:

- Maximum operating altitude; and,
- Maximum takeoff and landing altitude.

(B) Maximum operating altitude, is an operating limitation required by § 29.1527 and delineates the maximum altitude up to which operation is allowed. This altitude normally constitutes the maximum cruise or enroute altitude.

(C) Maximum weight, altitude and temperature for takeoff and landing constitutes a limitation. The maximum takeoff and landing altitude may be coincident with but never above the maximum operating altitude limitation. Takeoff and landing and hover ceiling data and presentation requirements are presented in §§ 29.51, 29.53 , 29.59, 29.63, 29.73, 29.1583 and 29.1587.

(ii) Procedures.

(A) In establishing the maximum takeoff and landing altitude, the following tests are normally required:

(1) Takeoff (§§ 29.51-29.63)

(2) Climb (§§ 29.64-29.67)

(3) Performance at minimum operating speed (§ 29.49)

(4) Landing (§ 29.75)

(5) Limiting height-speed envelope (§ 29.87)

(6) IGE controllability (§ 29.143c)

(7) Cooling (§§ 29.1041-29.1045)

(8) Engine operating characteristics (§ 29.939)

Specific guidance on test methodology and data requirements is provided in applicable paragraphs of this AC.

(B) As detailed in subparagraph b(2) above, the maximum allowable extrapolation of H-V, IGE controllability and engine operating characteristics is ±2,000 feet. Therefore, the maximum takeoff and landing altitude presented in the RFM is not normally more than 2,000 feet above the density altitude experienced at the high altitude test site.

(C) Prior to Amendment 29-21, H-V information was an operating limitation. With the adoption of Amendment 29-21, the H-V curve is performance information for Category B rotorcraft with nine or less passenger seats but remains a limitation for Category A rotorcraft and Category B rotorcraft with 10 or more passenger seats.

(D) Prior to Amendment 29-24, IGE controllability was required in 17 knots of wind to the maximum takeoff and landing conditions. With the adoption of Amendment 29-24, if IGE or OGE hover performance is presented for a Category B rotorcraft to an altitude in excess of that for which IGE controllability at 17 knots is presented, the maximum safe wind demonstrated for hover operations must be presented in the RFM. The amendment did not change the requirement for Category A rotorcraft.

(E) The requirements for data collection and presentation in the RFM vary depending upon the certification basis of the rotorcraft. These requirements are presented by regulation and amendment in figures AC 29.45-1 and AC 29.45-2.

(F) The maximum takeoff and landing altitude may be extrapolated no greater than the values given in paragraph b(2) above and not above the lowest limiting altitude resulting from the requirements listed in subparagraph A of this paragraph.

(4) Temperature Effects.

(i) Background.

(A) The regulations prohibit any unsafe design feature throughout the range of environmental conditions for which certification is requested. The regulations also require that the performance and handling qualities be determined over the approved range of atmospheric variables selected by the applicant.

(B) Substantiation of temperature effects on performance and handling characteristics is required throughout the approved temperature range. In the past, approved analyses were frequently accepted for determining the extreme temperature effects on performance and flight characteristics. With the introduction of newer, higher performance rotorcraft, advanced rotor blade designs, higher airspeeds, and blade mach numbers, the previous methods have proven to be insufficient. Therefore, the performance and flight characteristics should be validated at extreme temperatures; however, analysis may be permitted if a suitable methodology is demonstrated.

(C) Various FAA/AUTHORITY cold weather programs have verified that rotorcraft can be affected, sometimes significantly, in both the performance and flying qualities areas. Hot temperature conditions although not shown to be as critical should be given consideration.

(D) Additionally, design deficiencies surfaced when the rotorcraft were exposed to temperature extremes and some of these difficulties were severe enough to require the redesign of equipment and/or materials. Therefore, to satisfy § 29.1309(a), the applicant needs to substantiate the total rotorcraft at the extreme temperatures for which certification is requested.

(ii) Procedures.

(A) The FAA/AUTHORITY is responsible for verifying the applicant's predictions of performance and handling characteristics at the temperature extremes for which certification is requested. A limited flight verification, if necessary, could include spot checks of hover and climb performance, IGE controllability, roughness determination, simulated power failure, static stability, height-velocity, V_{NE}/V_D evaluations, ground resonance, etc. In addition, systems should be evaluated to determine satisfactory operations.

(B) Extrapolation of test data should only be allowed if the applicant's predicted or calculated data is verified by actual test but in any case extreme caution should be used for extrapolations that are -10°C below or +20° C above those values tested.

(5) Weight Effects. Test weights should be maintained within +3 percent and -1 percent of the target weight for each data point. Weight may be extrapolated only along an established W/σ line within the allowable altitude extrapolation range.

(6) Engine Power - Turboshaft Engine.

(i) Background.

(A) The purpose of rotorcraft performance flight testing is to obtain accurate quantitative flight test performance data to provide flight manual information.

(B) Flight tests are designed to investigate the overall performance capabilities of the rotorcraft throughout its operating envelope. This testing furnishes information to be included in the flight manual and provides a means of validating the predicted performance of the rotorcraft with a minimum installed specification engine.

(C) The horsepower used to complete the flight manual performance must be based on horsepower values no greater than that available from the minimum uninstalled specification engine after it is corrected for installation losses. A minimum uninstalled specification engine is one that, on a test stand under conditions specified by the engine manufacturer, will produce the certificated horsepower values at specification temperatures and/or speeds. The specification values may be either a rating or limit. Some engine manufacturers certify an engine to a specified horsepower at a particular engine temperature or speed rating with higher allowable limits. The limit is the maximum value the installed engine is allowed in order to develop the specification horsepower. Prior to installation of each engine in a rotorcraft, the performance is measured by the engine manufacturer. This is done by making a static test run in a test cell and referring the results to standard day, sea level conditions. The performance parameters obtained are presented as uninstalled engine characteristics on a test log sheet. This is commonly referred to as a “final run sheet.” Figure AC 29.45-3 compares a typical engine to one the manufacturer has certified as a minimum uninstalled certified engine.

(D) After engine certification, the engine manufacturer is responsible to ascertain that each engine delivered will produce, as a minimum, the certified horsepower values without exceeding specification operating values; therefore, a “final run sheet” is created for every engine produced. Additionally, if needed, arrangements can usually be made with the engine manufacturer to obtain a torque system calibration for individual engines. This will further optimize the accuracy of the engines used in the flight test program. The engine manufacturer will also provide predicted uninstalled power available for the various power ratings. This information may be derived from an engine computer “card deck” and from charts and tables in the engine detail installation manual. These data also provide engine performance for the range of altitudes and temperatures approved for the engine and include methods for correcting this performance for installation effects. The parameters contained in a typical “card deck” are plotted for one engine rating in figure AC 29.45-4.

(E) Several power losses may be associated with installing an engine in a rotorcraft. Typical losses are air inlet losses, gear losses, air exhaust losses, and powered accessory losses such as electrical generators. Additional flight manual performance considerations are the torque indicating system accuracy and torque needle split. The predicted uninstalled power available engine characteristics cannot be assumed to be the actual power available after the engine is installed in the rotorcraft because this procedure would neglect the installation power losses. It is necessary to know the installation losses in order to determine the flight manual performance. Installation losses are reflected reductions in available horsepower resulting from being installed in a rotorcraft. These losses usually consist of those incurred due to engine inlet and/or exhaust design. The rotorcraft manufacturer usually conducts test to confirm the installed specification. Methods used vary widely between manufacturers, but usually include some combination of ground and flight tests. Figure AC 29.45-5 is a typical example of an installed power available chart for one set of conditions.

(F) This predicted installed power available is, in most cases, lower than obtained on a test stand. This is especially true at lower airspeeds where exhaust reingestion decreases the available horsepower output and changes in airflow routing. The rotorcraft manufacturer may elect to determine the installation losses for different flight conditions to take any airspeed advantages. This is acceptable if, for example, the hover performance is based on the actual horsepower available from a minimum installed specification engine in a hover. Likewise, it is permissible for the rotorcraft manufacturer to determine his climb performance based on the actual horsepower available from a minimum installed specification engine at the published climb airspeed. This will allow the manufacturer to take advantage of, for example, increased inlet efficiency.

(ii) Procedures.

(A) To this point the minimum installed specification engine horsepower output has been predicted and calculated for various flight conditions. It is imperative that the predicted values be verified by actual flight test. The flight test involves obtaining engine performance measurements at various power settings, altitudes, and ambient temperatures. The data should be obtained at the actual flight condition for which the performance is to be presented (i.e., hover, climb, or cruise).

(B) Following an initial application of power, engine temperature and/or RPM can significantly decrease for a period of time as torque is held constant. Said another way, torque will increase if RPM and/or temperature is held constant. This is a characteristic typical of turbine engines due largely to expansion of turbine blades and reduced clearances in the engine. Some engines may show a temperature increase at constant power due to engine or temperature sensing system peculiarities. An engine will usually establish a stabilized relationship of power parameters in approximately 2 or 3 minutes. For this reason, the following procedure should be used when obtaining in-flight engine data.

(1) To determine the applicable value (takeoff, 30-second, and 2 1/2-minute power), the engine is first stabilized at a low power setting. After stabilization, rapidly increase the power demand to takeoff, 30-second and 2 1/2-minute power levels as necessary. Record the engine parameters as soon as the specification torque, temperature, or speed is attained. Care must be taken not to exceed a limit. These readings should be obtained approximately 15 seconds after power is initially applied.

(2) To determine the 30-minute and/or maximum continuous power values, approximately 2 to 3 minutes of stabilization time is generally used, but up to 5 minutes stabilization time is allowed. The reason for the different procedures is when a pilot requires takeoff or 2 1/2-minute power values he is in a critical flight condition and does not have the luxury of waiting for the engine(s) to produce rated power. Stabilization time is allowed for the maximum continuous and 30-minute ratings

because these values are not associated with flight conditions for which power is needed immediately. An engine may be certified to produce a specification horsepower at a particular temperature or engine speed rating with higher maximum limit value approved. Only the rating values should be used to determine the installation losses. The limit values of engine temperature and/or speed are established and certified to allow specification powers to continue to be developed as the engine deteriorates in service.

(C) The in-flight measurements recorded with the engine(s) on the flight test rotorcraft must be corrected downward if the test engine is above minimum specification and corrected upward for a test engine that is below minimum specification. This correction is necessary to verify that a minimum installed specification engine installed on a production rotorcraft is capable of producing the horsepower values used to compute the flight manual performance without exceeding any engine limit. In addition, if the production rotorcraft's power measurement devices have significant (greater than 3 percent) power error, this error must be accounted for in a conservative manner.

(D) On multiengine rotorcraft, the engine location may result in different installation losses between engines. If this condition exists, multiengine performance should be based on a total of the different minimum installed specification horsepower values. One engine inoperative performance must be based on the loss of the engine which has the lowest installation losses. Additionally, the power losses due to such items as accessory bleed air, particle separators, etc., must be accounted for accordingly.

(E) Power available data should be obtained throughout the test program at various ambient conditions. Some engines have devices which restrict the mechanical N_G speed to a constant corrected speed at cold temperatures. Others may limit power to a minimum fuel flow value which would be encountered only at certain ambients. Others may limit by torque limiting devices. Therefore, power available data should be obtained at various ambients to verify that all limiting devices are functioning properly and have not been affected by the installation.

(F) Through use, turbine engine power capabilities decrease with time. This is called engine deterioration. Deterioration is largely a function of the particular engine design, and the manner and the environment in which the engine is operated. There is a need, therefore, to provide a method which can be used in service to periodically determine the level of engine deterioration. A power assurance curve is usually provided to allow the flightcrew to know the power producing capabilities of any engine. A power assurance check is a check of the engine(s) which will determine that the engine(s) can produce the power required to achieve flight manual performance. This check does not have to be done at maximum engine power. Figure AC 29.45-6 is a typical power assurance curve for an installed engine showing minimum acceptable torque which assures that power is available to meet the rotorcraft flight manual performance. Some power assurance curves have maximum allowable N_G limits that must not be exceeded for a given torque value. An in-flight power assurance check may be used in addition to the pre-takeoff check. The validation of either check must be done by the methodology used to determine the installed minimum specification engine power available. For the in-flight power assurance check there must be full accountability for increased efficiency due to such items as inlet ram recovery, absence of exhaust reingestion, etc. A power assurance check done statically and one conducted in-flight must yield the same torque margin(s). An engine may pass power assurance at low power but still may not be capable of producing the rated horsepower values. This occurs when the curve of measured corrected horsepower and corrected temperature for the engine intersects the minimum uninstalled specification engine curve. If this condition exists, the entire power assurance and power available information may need to be reestablished.

(7) Deteriorated Engine Power - Turboshaft Engine.

(i) Background.

(A) A specific engine model may have been certificated for operation with power which has "normally" deteriorated below specification. This "normal" deterioration refers to a gradual loss in engine performance, possibly caused by compressor erosion, as opposed to a sudden performance loss which may be due to mechanical damage. The application for deteriorated engine power should not be confused with the installed mechanical engine derating which is frequently used to match transmission and engine power capabilities.

(B) The use of deteriorated power is intended to allow continued operations with an engine which is serviceable and structurally sound, although aircraft performance may be depreciated. The useful life of the engine may, therefore, be extended at a dollar savings to the operator.

(C) Although installed performance is the primary topic in this discussion, considerations must be given to other operational characteristics and systems which may be affected by depreciated engine power. These include:

(1) Engine characteristics (§ 29.939). The reduced compressor discharge pressure, P_C, would reduce engine surge margin and possibly affect engine response and engine air-restart capability. These items should be addressed, but flight testing may not be required depending on the individual engine/aircraft installation and fuel scheduling mechanism.

(2) Performance of customer bleed air systems may be degraded slightly. No problem would be anticipated unless certain items within the system depend on a critical P_C for their function.

(3) The maximum attainable gas producers speed, and thus power available under certain ambients, may be affected if P_C pressure is an input to the fuel scheduling mechanism.

(4) Systems for surge protection which schedule on P_C pressure such as bleed valves, flow fences, bleed bands, and variable inlet guide vanes may be influenced. The affect would normally be negligible unless when installed, the installation losses combined with reduced P_C because of deterioration, would cause the bleed device to open and reduce power at any one of the engine ratings.

(ii) Procedures.

(A) The need for flight tests to verify predicted power available with deteriorated engines depends on the scope of testing which occurred during initial certification. If the original rotorcraft certification included flight testing as described in paragraph (6) (engine power-turboshaft engines) herein for validation of power available, the need for a demonstration with deteriorated engines, is greatly diminished and perhaps eliminated.

(B) If flight testing to verify deteriorated engine power available is deemed necessary, the procedure used would be the same as that described in paragraph (6) (engine power-turboshaft engines), except that the data would be corrected downward to a deteriorated engine runline. Efforts should concentrate on obtaining data in areas of the operational envelope where maximum gas producer speed is likely to be attained, or where bleed valves or other devices which schedule on gas producer discharge pressure are likely to function. On many installations maximum gas producer speed will occur cold and high; bleed valves and other devices which schedule on gas producer discharge pressure are most likely to function and reduce power on a hot day at low altitude.

(C) The adjustments to the normal power assurance check procedures for deteriorated engines will be influenced by the preferences of the aircraft manufacturer and by any special stipulations of the engine certification region established as a condition for the engine to remain in service when below specification. Possibly, more stringent and more complicated procedures will be introduced for deteriorated power; for example, an in-flight trend monitoring program with the associated bookkeeping duties may be required. Such an in-flight procedure must be evaluated by flight tests as described in paragraph (6) (engine power-turboshaft engines) herein. Normally, however, the manufacturer would be expected to present a modification, or extension of the power assurance procedure already in place for the specification engine, which could eliminate the need for flight test evaluation.

(D) If a complex power assurance procedure is presented with involved data reduction and trending requirements, consideration should be given to restricting the use of deteriorated power to operators where close control over operations is exercised and/or the operator has demonstrated his ability to operate safely with deteriorated engines.

AC 29.45A. § 29.45 (Amendment 29-24) PERFORMANCE - GENERAL.

a. Explanation. Amendment 29-24 adds § 29.45(f) to the regulation. This section establishes the requirement for furnishing power assurance information for turbine powered aircraft. This information is to provide the pilot a means of determining, prior to takeoff, that each engine will produce the power necessary to achieve the performance presented in the rotorcraft flight manual (RFM).

b. Procedures. All of the policy material pertaining to this section remains in effect. In addition, the power assurance information included in the RFM should be verified. Although this requirement is normally met with a power assurance curve, other methods of compliance may be proposed.

AC 29.45B. § 29.45 (Amendment 29-24) PERFORMANCE - GENERAL.

a. Explanation. Amendment 29-34 added the requirements for certification of 30-second/2-minute One Engine Inoperative (OEI) power ratings. For rotorcraft approved for the use of 30-second/2-minute OEI, partial power checks currently accomplished with approved power assurance procedures for lower power levels may not be sufficient to guarantee the ability to achieve the 30-second power level.

b. Procedures. Information provided in paragraph AC 29 MG 9 includes guidance material on power assurance procedures to ensure that the OEI power level can be achieved. The guidance material presented in paragraphs AC 29.45 and AC 29.45A continue to apply.

CERTIFICATION BASIS

		FAR 29			CAR 7
Requirements		29-Amdt. 21	29-Amdt. 1	Original	Original
H-V Ref. 29.25 29.87 29.1517 29.1581 29.1583 7.11 7.715 7.741	**CAT A TEST CONDITIONS**	Cat A & B (>9 pax seats): 1. W.A.T. for which t.o. and ldg. are approved. 2. Failure of critical engine.	Cat A: 1. MGW Sea Level 2. Max. IGE wt. Max. alt. capability. 3. Failure of critical engine.	Cat A: 1. MGW Sea Level 2. Max. IGE wt. Max. alt. capability. 3. Failure of critical engine.	Cat. A: 1. MGW Sea Level 2. Max. IGE wt. Max. alt. Capability. 3. Failure of critical engine.
AC 29-2C paras AC 29.45 & AC 29.79	**CAT A RFM**	3. H-V is limitation. 4. Type of ldg. surface.	4. H-V is limitation. 5. Type of ldg. Surface.	4. H-V is limitation. 5. Type of ldg. surface.	4. H-V is limitation. 5. Type of ldg. surface.
	CAT B TEST CONDITIONS	Cat B (</=9 pax seats): 1. MGW Sea Level 2. Max. OGE wt. Max. alt. Capability 3. Complete power failure, or failure of critical engine (w/eng isolation).	Cat B: 1. MGW Sea Level 2. Max. IGE wt. Max. alt. Capability 3. Complete power failure, or failure of critical engine (w/eng isolation).	Cat B: 1. MGW Sea Level 2. Max. IGE wt. Max. alt. Capability 3. Complete power failure, or failure of critical engine (w/eng isolation.)	Cat B: 1. MGW Sea Level 2. Max. IGE wt. Max. alt. Capability 3. Complete power failure, or failure of critical engine (w/eng isolation.)
	CAT B RFM	4. H-V is perf. Info. 5. Type of ldg. surface.	4. H-V is limitation info. 5. Type of ldg. Surface.	4. H-V is limitation info. 5. Type of ldg. surface.	4. H-V is limitation info. 5. Type of ldg. surface.

FIGURE AC 29.45-1. H-V REQUIREMENTS

CERTIFICATION BASIS

		FAR 29			CAR 7
Requirements		29-Amdt. 24	29-Amdt. 3	Original	Original
IGE CONTROL Ref. 29.25 29.1583 7.121 7.743	**CAT A TEST CONDITIONS**	Cat A 1. W.A.T. for which t.o. and ldg. are approved. 2. Critical wt. Critical CG Critical Nr 3. Wind not less than 17 kts.	1. Conditions selected by the applicant. 2. Critical CG Critical Nr 3. Wind not less than 17 kts.	1. Conditions selected by the applicant. 2. Critical CG Critical Nr 3. Wind not less than 20 mph.	1. Conditions selected by the applicant. 2. Critical CG Critical NR 3. Wind not less than 20 mph.
AC 29-2C paras AC 29.45 and AC 29.143	**CAT A RFM**	4. Max. allowable wind is limitation.	4. Max safe wind above max. alt. For which 17 kt. Wind envelope is established is perf. info.	4. Max safe wind above alt. for which 17 kt. wind envelope is established is perf. info.	4. Max. allowable wind above the altitude for which 20 mph wind envelope is est. is perf. info.
	CAT B TEST CONDITIONS	Cat B: 1. W.A.T. for which t.o. and ldg. are approved. 2. Critical wt. Critical CG Critical Nr 3. Wind speed & quad selected by the applicant.			
	CAT B RFM	4. Max. safe wind is perf. info.			

FIGURE AC 29.45-2 IGE CONTROLLABILITY REQUIREMENTS

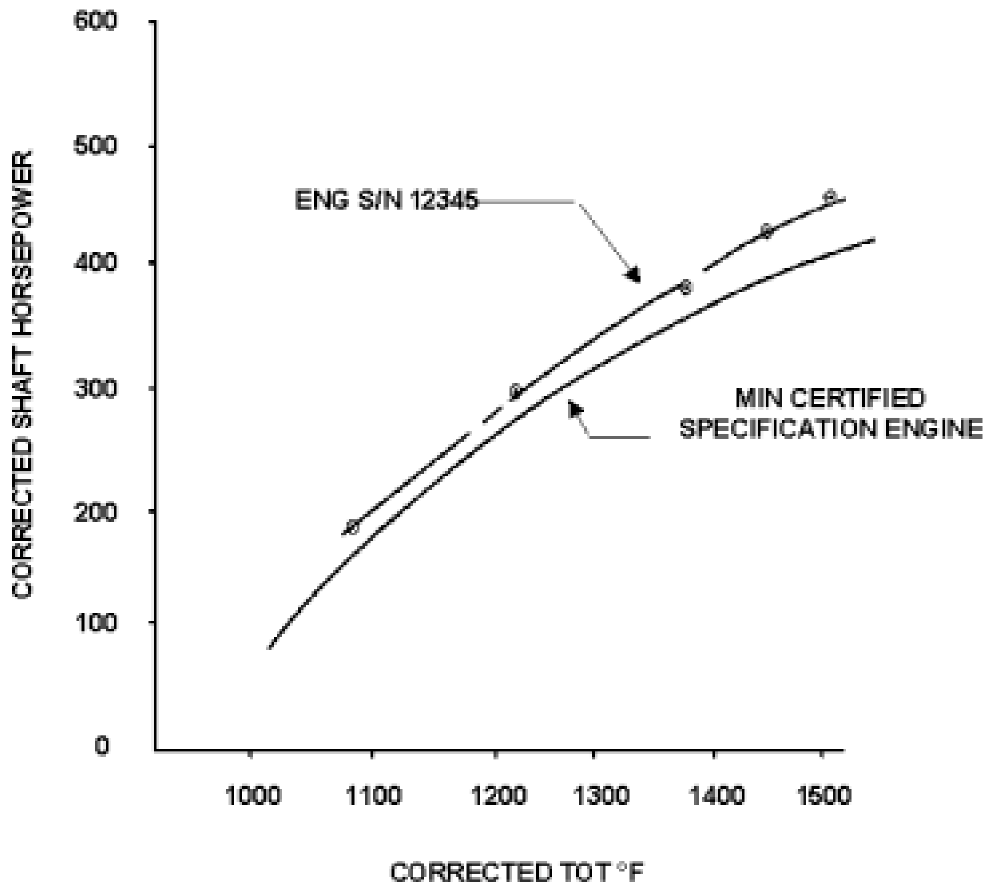


FIGURE AC 29.45-3 SHAFT HORSEPOWER VS TURBINE OUTLET TEMPERATURE - SEA LEVEL STANDARD DAY

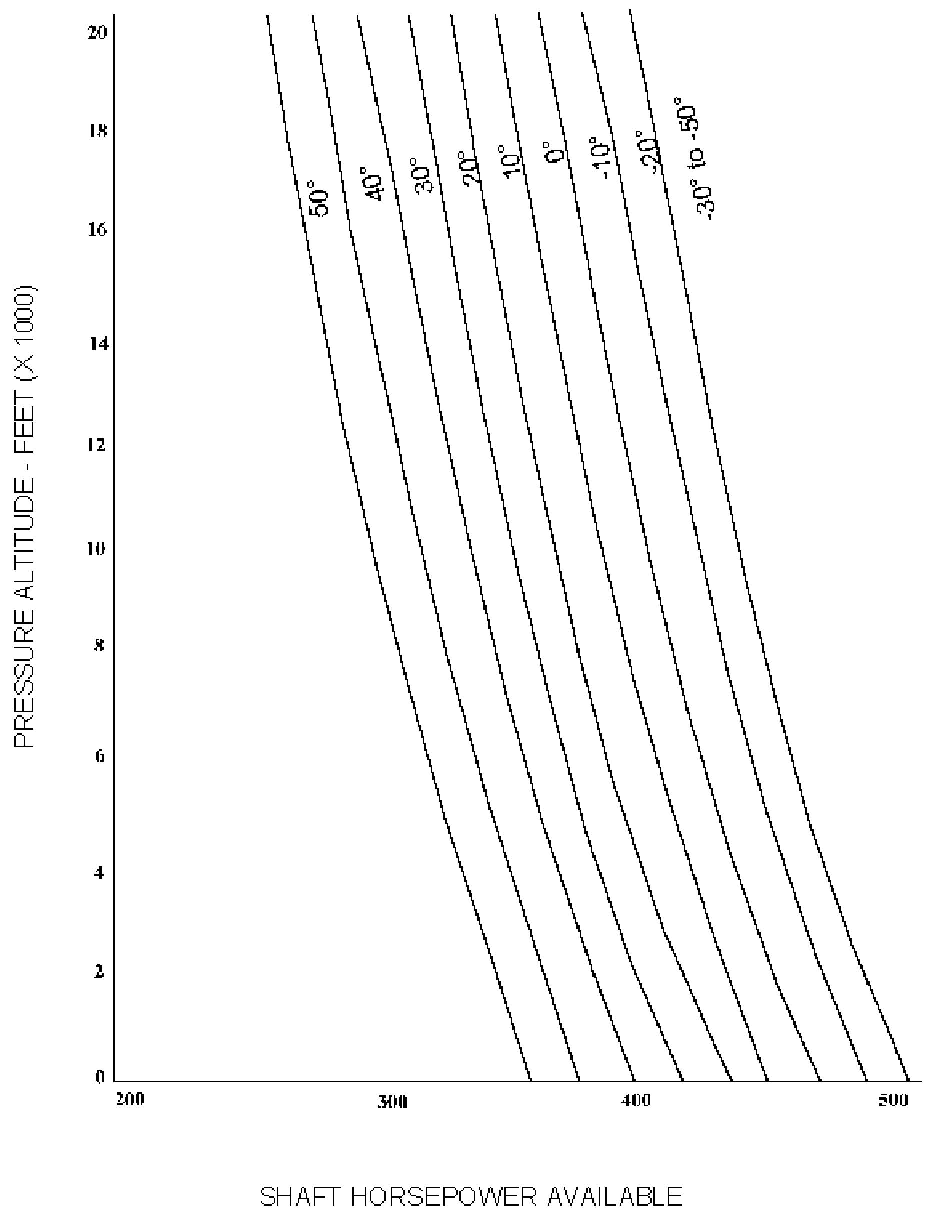


FIGURE AC 29.45-4 UNINSTALLED TAKEOFF POWER AVAILABLE

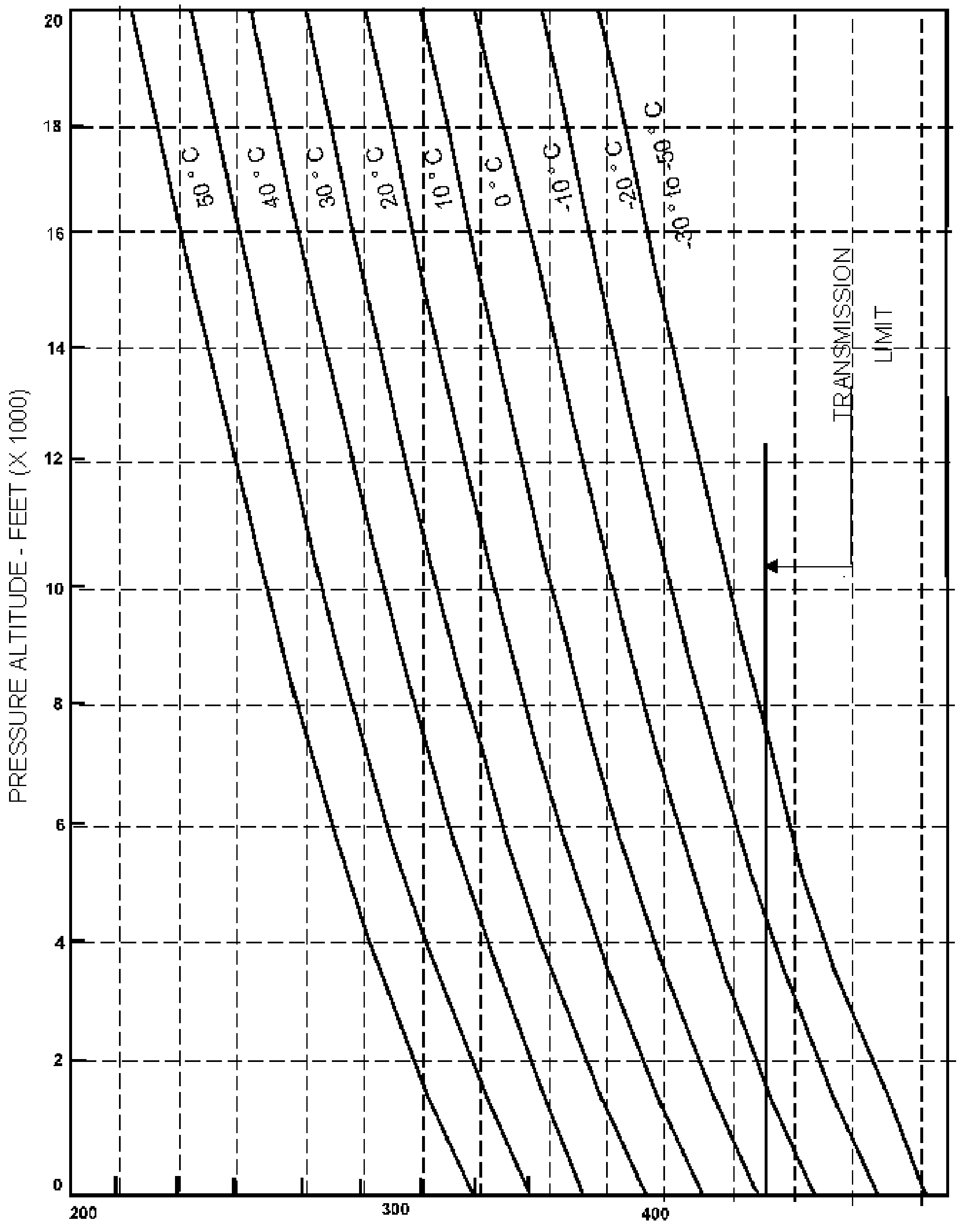


FIGURE AC 29.45-5 INSTALLED TAKEOFF POWER AVAILABLE, ANTI-ICE OFF, 400 RPM

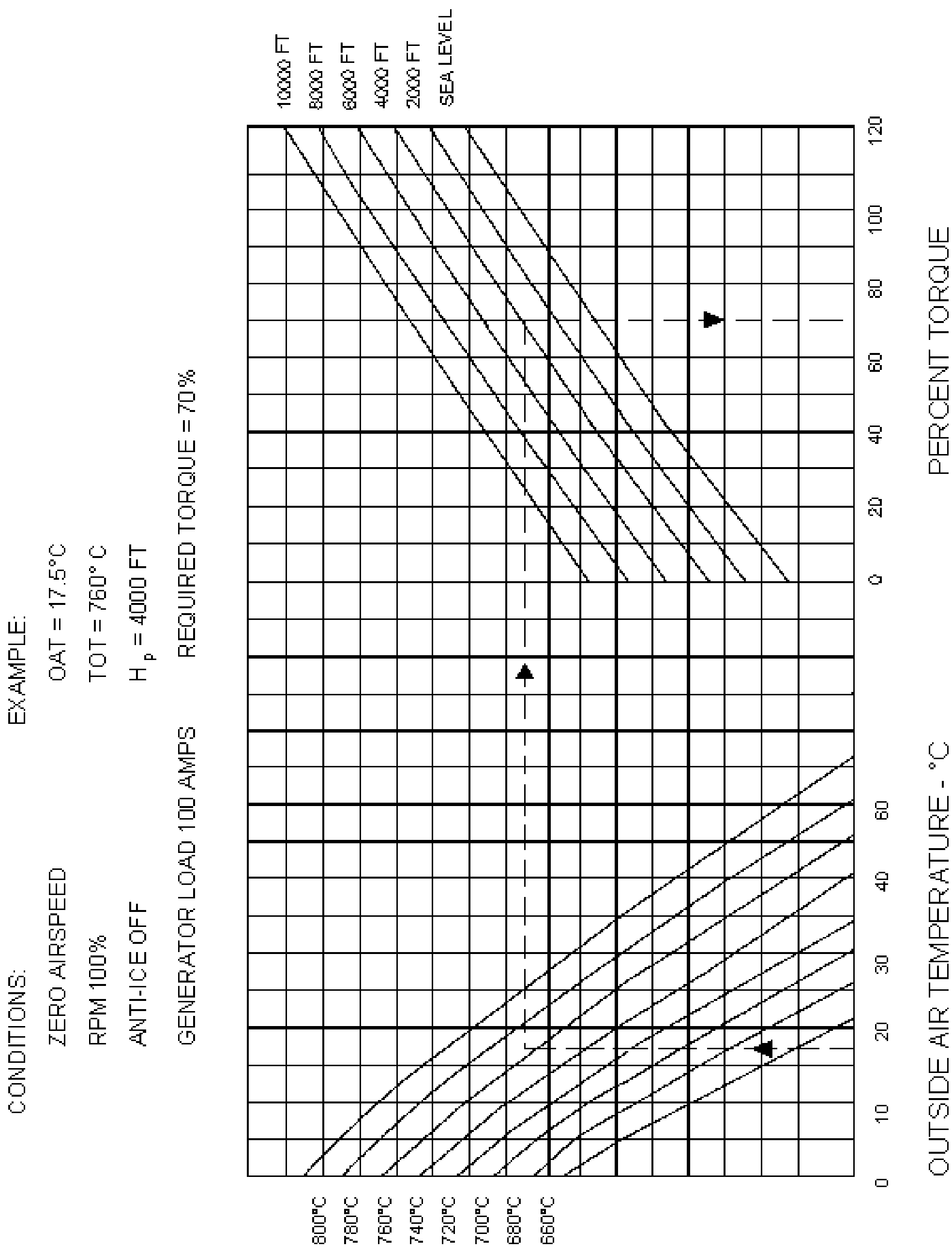


FIGURE 29.45-6 POWER ASSURANCE CHECK CHART

AC 29.49. § 29.49 (Amendment 29-39) PERFORMANCE AT MINIMUM OPERATING SPEED. HOVER PERFORMANCE FOR ROTORCRAFT.

(For performance at minimum operating speed and for hover performance prior to Amendment 39, see § 29.73 and paragraph AC 29.73.)

a. Explanation.

(1) Amendment 29-39 redesignated § 29.73 as § 29.49 to relocate the requirements for rotorcraft hover performance. For the purpose of this manual, the word "hover" applies to a rotorcraft that is airborne at a given altitude over a fixed geographical point regardless of wind. Pure hover is accomplished only in still air.

(2) Under § 29.49, hover performance should be determined at a height consistent with the takeoff procedure for Category A rotorcraft and IGE for Category B rotorcraft. Additionally, OGE hover performance should be determined for both Category A and B rotorcraft. Hover OGE is that condition, where an increase in height above the ground will not require additional power to hover. Hover OGE is the absence of measurable ground effect. It can be less than one rotor diameter at low gross weight increasing significantly at high gross weights. The lowest OGE hover height at gross weight may be approximated by placing the lowest part of the vehicle 1 ½ rotor diameters above the surface.

(3) The objective of hover performance tests is to determine the power required to hover at different gross weights, ambient temperatures, and pressure altitudes. Using nondimensional power coefficients (C_p) and thrust coefficients (C_t) for normalizing and presenting test results, a minimum amount of data are required to cover the rotorcraft's performance operating envelope.

(4) Hover performance tests must be conducted over a sufficient range of pressure altitudes and weights to cover the approved ranges of those variables for takeoff and landings. Additional data should be acquired during cold ambient temperatures, especially at high altitudes, to account for possible Mach effects.

(5) The minimum hover height for which data should be obtained and subsequently presented in the flight manual should be the same height consistent with the minimum hover height demonstrated during the takeoff tests. Refer to paragraph AC 29.51 for the procedure to determine the minimum allowable hover height.

b. Procedures.

(1) Two methods of acquiring hover performance data are the tethered and free flight techniques. The tethered technique is accomplished by tethering the rotorcraft to the ground using a cable and load cell. The load cell and cable are attached to the ground tie-down and to the rotorcraft cargo hook. The load cell is used to measure the

rotorcraft's pull on the cable. Hover heights are based on skid or wheel height above the ground. During tethered hover tests, the rotorcraft should be at light gross weight. The rotorcraft will be stabilized at a fixed power setting and rotor speed at the appropriate skid or wheel height. Once the required data are obtained, power should be varied from the minimum to the maximum allowed at various rotor RPM. This technique will produce a large C_t/C_p spread. The load cell reading is recorded for each stabilized point. The total thrust the rotor produces is the rotorcraft's gross weight, weight of the cables and load cell plus cable tension. Care must be taken that the cable tension does not exceed the cargo hook limit or load capacity of the tie-down. For some rotorcraft, it may be necessary to ballast the rotorcraft to a heavy weight in order to record high power hover data.

(2) The pilot maintains the rotorcraft in position so that the cable and load cell are perpendicular to the ground. To insure the cable is vertical, two outside observers, one forward of the rotorcraft and one to one side, can be used. Either hand signals or radio can be used to direct the pilot. The observers should be provided with protective equipment. This can also be accomplished by attaching two accelerometers to the load cell which sense movement along the longitudinal and lateral axes. Any displacement of the load cell will be reflected on instrumentation in the cockpit and by reference to this instrumentation, the rotorcraft can be maintained in the correct position. Accurate load cell values may also be obtained by measuring cable angles and, through geometry, determining a corrected load cell value. Increased caution should be utilized as tethered hover heights are decreased because the rotorcraft may become more difficult to control precisely. The tethered hover technique is especially useful for OGE hover performance data because the rotorcraft's internal weight is low and the cable and load cell can be jettisoned in the event of an engine failure or other emergency.

(3) To obtain consistent data, the wind velocity should be 3 knots or less. Large rotorcraft with high downwash velocities may tolerate higher wind velocities. The parameters usually recorded at each stabilized condition are:

(i) Engine torques.

(ii) Rotor speed.

(iii) Ambient temperatures.

(iv) Pressure altitude.

(v) Fuel used (or remaining).

(vi) Load cell reading.

(vii) Generator(s) load.

As a technique, it is recommended the rotorcraft be loaded to a center of gravity near the hook to minimize fuselage angle changes with varying powers. All tethered hover data should be verified by a limited spotcheck using the free flight technique. The free flight technique as contained in paragraph b(4) below will determine if any problems, such as load cell malfunctions have occurred. The free flight hover data must fall within the allowable scatter of the tethered data.

(4) If there are no provisions or equipment to conduct tethered hover tests, the free flight technique is also a valid method. The disadvantage of this technique as the primary source of data acquisition is that it is very time consuming. In addition a certain element of safety is lost OGE in the event of emergency. The rotorcraft must be reballasted to different weights to allow the maximum C_t/C_p spread. When using the free flight technique, either as a primary data source or to substantiate the tethered technique, the same considerations for wind, recorded parameters, etc., as used in the tethered technique apply. Free flight hover tests should be conducted at CG extremes to verify any CG effects. If the rotorcraft has any stability augmentation system which may influence hover performance, it must be accounted for.

(5) It is extremely difficult to determine when a rotorcraft is hovering OGE at high altitudes above ground level since there is no ground reference. In a true hover, the rotorcraft will drift with the wind. Numerous techniques have been tried to allow OGE hover data acquisition at high altitudes, all of which have resulted in much data scatter. Until a method is proposed and found acceptable to the FAA/AUTHORITY, OGE hover data must be obtained at the various altitude sites where IGE hover data is obtained. Hover performance can usually be extrapolated up to a maximum of 4,000 feet.

AC 29.51. § 29.51 TAKEOFF DATA - GENERAL.

a. Explanation. Section 29.51 details the conditions under which takeoff performance data can be obtained and presented in the FAA/AUTHORITY approved flight manual. The flight manual must also contain the technique(s) to be used to obtain the published flight manual takeoff performance. Technique should not be confused with exceptional pilot skill and/or alertness as mentioned in § 29.51. Rotorcraft are different from one another and due to this, different pilot techniques are sometimes required to achieve the safest and most optimum takeoff performance. The recommended technique that is published in the flight manual and used to achieve the performance must be determined to be one that the operational pilot can duplicate using the minimum amount of type design cockpit instrumentation and the minimum crew.

b. Background.

(1) Certain special takeoff techniques are necessary when a rotorcraft is unable to takeoff vertically because of altitude, weight, power effects, or operational limitations. The recommended technique used to take off under such conditions is to accelerate the rotorcraft in-ground-effect (IGE) to a predetermined airspeed prior to climbout. Takeoff

tests are performed to determine the best repeatable technique(s) for a particular rotorcraft over the range of weight, altitude, and temperature for which certification is requested.

(2) The primary factor which determines the rotorcraft's takeoff performance is the amount of excess power available. Excess power available is the difference between the power required to hover at the reference height above the ground and the takeoff power available from a minimum installed specification engine. Utilizing the total power available to execute a takeoff may not be operationally feasible due to such items as HV constraints. In such situations, hover power required plus some power increment may be the maximum that can be used and the resulting performance determined accordingly.

(3) Landing gear height above the ground should not be greater than that demonstrated satisfactorily for HV, rejected takeoff, and that height for which IGE hover performance data is presented in the RFM, or less than that height below which ground contact may occur when accomplishing takeoff procedures. For rotorcraft fitted with wheels, a running takeoff procedure may be accepted. The hover reference height is established as the minimum landing gear height above the takeoff surface, from which a takeoff can be accomplished consistently in zero wind without contacting the runway. Category B takeoff must be accomplished with power fixed at the power required to hover at the reference height (not greater than the height for which IGE performance data is presented).

c. Procedure. There are different techniques which may be used in order to determine which method is best for a particular rotorcraft. The most commonly accepted method is the hover and level acceleration technique. In this technique, the rotorcraft is stabilized in a hover at the reference height. From the stabilized hover, the rotorcraft is accelerated to the climbout airspeed using the predetermined takeoff power. When the desired climbout airspeed is achieved, the rotorcraft is rotated and the climbout is accomplished at the schedule airspeed(s) and constant rotor RPM. Power adjustments may be accomplished to maintain targeted power except where procedure requires high workload outside cockpit (i.e., that portion of takeoff where horizontal acceleration close to the ground has pilot scan outside the cockpit and adjustment of engine torque or temperature would require an undue increase in workload).

AC 29.51A. § 29.51 (Amendment 29-39) TAKEOFF DATA - GENERAL.

a. Explanation. Amendment 39 added takeoff requirements in new §§ 29.55, 29.60, 29.61 and 29.62.

b. Procedures. The guidance material presented in paragraph AC 29.51 continues to apply.

AC 29.53. § 29.53 TAKEOFF: CATEGORY A.

a. Explanation.

(1) A Category A takeoff typically begins with an acceleration and/or climb from a hover to a critical decision point. The rule requires that the critical decision point (CDP) be defined for the pilot in terms of an indicated altitude and airspeed combination. However, other parameters to define the CDP have been accepted by the FAA/AUTHORITY on an equivalent safety basis. A regulatory project has been established to change the rule permitting other parameters to be used for CDP definition.

(2) The requirement to define CDP as a combination of both airspeed and height above the takeoff surface is based on a minimum required total energy concept. A specific minimum combination of kinetic energy (airspeed) and potential energy (height) must be attained at the CDP to be assured that a continued takeoff can be accomplished following the complete failure of one engine. In § 29.53(b), CDP is required to be "…a combination of height and speed selected by the applicant…" Any other method proposed to define CDP must provide the same level of safety as would be obtained using an airspeed-height combination. When using "time," "height," or "airspeed" only as alternative methods of identifying the CDP, they must be combined with a precisely defined takeoff path and crew procedure in order to provide the required equivalent level of safety. In addition, it must be demonstrated that the pilot technique used during the takeoff sequence is easily repeatable and consistently produces the required energy (i.e., airspeed and altitude combination) when the CDP "time," "height," or "airspeed" is attained. This condition should be verified during the flight test program.

(3) If an engine fails at the CDP or at any point in the takeoff profile prior to attaining CDP, the rotorcraft must be able to land safely within the established rejected takeoff distance. Flight testing to determine the Category A rejected takeoff distance is very similar to height-velocity testing and should be approached with caution. The initial Category A takeoff profiles should be outside of the Category B height-velocity envelope. Previous programs have shown the low speed point immediately after application of power to be particularly critical.

(4) If an engine fails at the CDP or at any subsequent point in the Category A takeoff profile, a continued safe climb-out capability is assured. The continued takeoff for conventional Category A runway profiles is designed to allow acquisition of the takeoff safety speed (V_{TOSS}), at a minimum of 35 feet above the takeoff surface and a positive rate of climb. During the continued takeoff profile, the pilot is assumed to be flying the rotorcraft via the primary flight controls (cyclic stick, collective, and directional pedals). Manipulation of the throttle controls or beep switches may be permitted as long as such manipulation can be accomplished readily by the pilot flying the rotorcraft without removing his hands from the cyclic and collective flight controls. These manipulations of engine controls should not make major adjustments in power, and

should not occur before attaining V_{TOSS}. In no case should this be less than 3 seconds after the critical engine is made inoperative.

(5) Both the rejected takeoff distance and the continued takeoff distance must be determined. Although 29.59(c) suggests a balanced field length requirement, this was not intended. Both rejected and continued takeoff distance should be included in the RFM performance with information stating that the longer distance determines the length of the required takeoff surface. Operations approvals can then determine the required takeoff surface (including stopways and clearways) appropriate for the specific operation.

(6) A typical Category A takeoff profile, assuming an engine failure at the CDP, is shown in figure AC 29.53-1.

b. Procedures.

None.

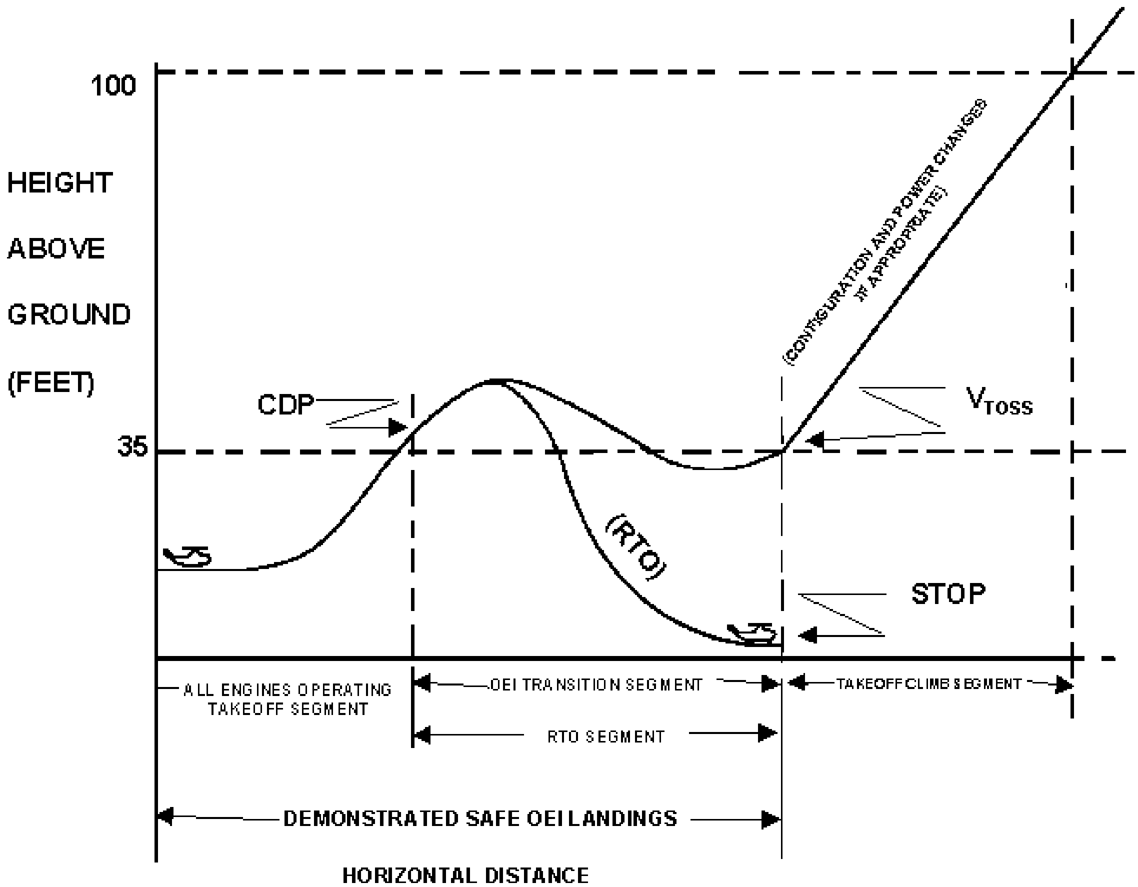


FIGURE AC 29.53-1 TAKEOFF PERFORMANCE CATEGORY A

AC 29.53A. § 29.53 (Amendment 29-39) TAKEOFF: CATEGORY A.

a. Explanation. Amendment 29-39 separated in the text, the Category A takeoff requirement from the definition of a decision point. Category A takeoff performance must be scheduled so that:

(1) If an engine failure is recognized at the Takeoff Decision Point (TDP) or at any point in the takeoff profile prior to attaining TDP, the rotorcraft must be able to land safely within the established rejected takeoff distance. Flight testing to determine the Category A rejected takeoff distance is very similar to height-velocity testing and should be approached with caution. The initial Category A takeoff profiles should be outside of the avoid area of the Category B height-velocity envelope. Previous programs have shown the low speed point immediately after application of power to be particularly critical.

(2) If an engine failure is recognized at the TDP or at any subsequent point in the Category A takeoff profile, a continued safe climb-out capability must be assured. The continued takeoff for conventional Category A runway profiles is designed to allow acquisition of the takeoff safety speed (V_{TOSS}) at a minimum of 35 feet above the takeoff surface and a positive rate of climb.

(3) Both the rejected takeoff distance and the continued takeoff distance should be determined. A balanced field length is not required by the regulation. Both rejected and continued takeoff distance should be included in the RFM performance section. Operations approvals can then determine the required takeoff surface (including stopways and clearways) appropriate for the specific operation.

(4) A typical Category A takeoff profile, assuming an engine failure prior to the TDP, is shown in figure AC 29.53A-1.

b. Procedures. None.

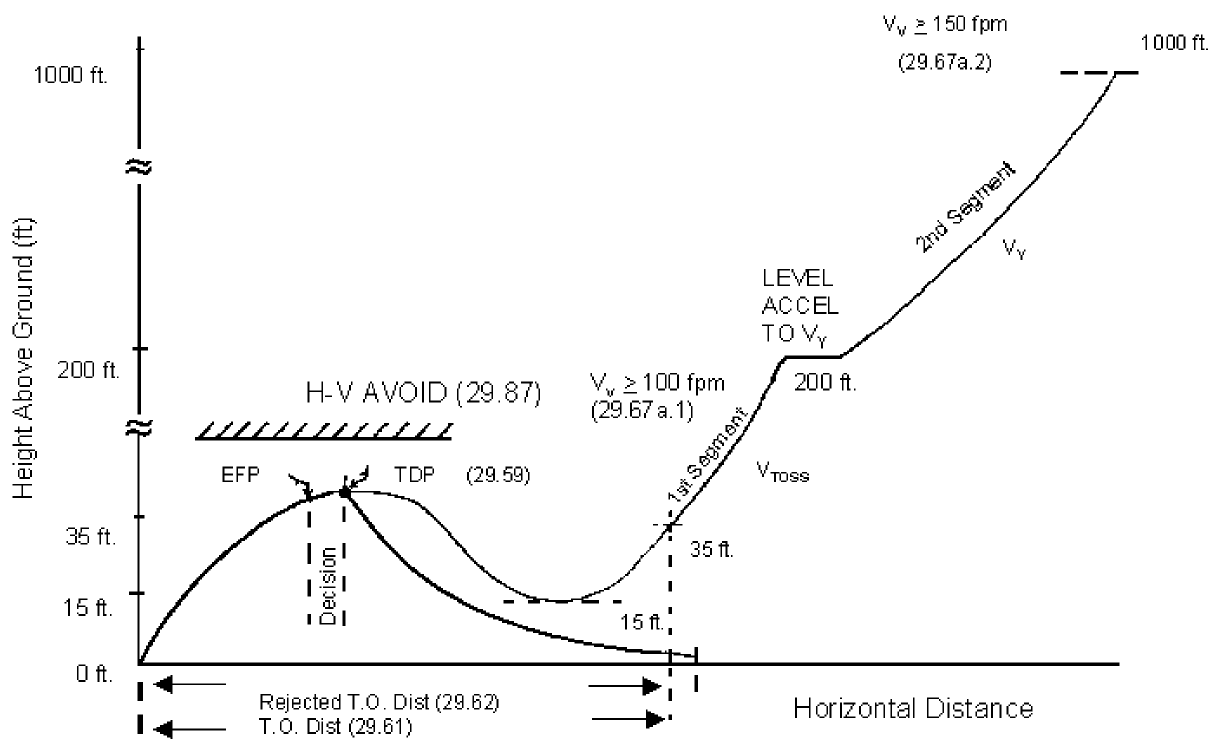


FIGURE AC 29.53A-1 TAKEOFF PERFORMANCE CATEGORY A

AC 29.55. § 29.55 (Amendment 29-39) TAKEOFF DECISION POINT: CATEGORY A.

a. Explanation.

(1) Amendment 29-39 added a new § 29.55 to redefine the TDP (previously called the CDP) and contained in § 29.53; it further removed the requirement to identify the TDP by height and airspeed, since height alone or other factors may be more appropriate. A Category A takeoff typically begins with an acceleration and/or climb from a hover to TDP. The rule requires that the TDP be defined for the pilot in terms of no more than two parameters such as an indicated height and airspeed combination.

(2) The definition of the TDP is based on a minimum required total energy concept. A specific minimum combination of kinetic energy (airspeed) and potential energy (height) should be attained at the TDP to ensure that a continued takeoff can be accomplished following the complete failure of one engine. In § 29.55(b), TDP is required to be defined by no more than two parameters. When using a single parameter such as time, height, or airspeed as a method of identifying the TDP, the identification must be combined with a precisely defined takeoff path and crew procedure to provide the required equivalent level of safety. In addition, it should be demonstrated that the pilot technique used during the takeoff sequence is easily repeatable and consistently produces the required energy (i.e., airspeed and height combination) when the TDP time, height, or airspeed is attained. This condition should be verified during the flight test program.

b. Procedures. None.

AC 29.59. § 29.59 (Amendment 29-24) TAKEOFF PATH: CATEGORY A.

a. Explanation. The Category A concept limits the rotorcraft takeoff weight such that if an engine failure occurs at or before the CDP, a safe landing can be made or if the engine fails at or after the CDP, the takeoff can be continued. The purpose of these tests is to define the CDP, evaluate the necessary pilot techniques, and determine the required takeoff area for either alternative. The condition of equal distances for either stopping or continuing the takeoff is called a "balanced" field length. The combination of altitude and speed at the CDP which produces a balanced field length is not required for certification. This section deals with the Category A takeoff and rejected takeoff profiles. The profiles necessarily involve consideration of an average pilot skill level as well as a sequence in which it is assumed various configuration adjustments are made to the rotorcraft.

(1) Takeoff. The Category A takeoff path begins with an all-engines-operating acceleration segment to the CDP and continues with a one-engine-inoperative acceleration to takeoff safety speed (V_{TOSS}). (See Conventional Takeoff Profile, figure AC 29.53-1, paragraph AC 29.53.) CDP is a "go/no-go" condition which is analogous to V_1 speed in transport airplanes. Prior to CDP the pilot is "stop" oriented,

and when an engine fails in this portion of the takeoff, he will abort because he has not yet achieved sufficient energy to assure continued flight. At the CDP the pilot becomes "go" oriented and when an engine fails at or beyond this point he will continue the takeoff because he no longer has sufficient surface area to abort the takeoff. The takeoff flight path and the CDP must be defined such that a safe landing can be made from any point up to the CDP. This profile may differ significantly from the takeoff flight path developed for Category B weights. The CDP is the last point in the takeoff profile at which a rejected takeoff capability within the scheduled takeoff surface distance is assured. If an engine failure does not occur, the pilot continues the climb and accelerates past the CDP to the recommended climb speed.

(2) Rejected Takeoff. The rejected takeoff profile begins with an all engine acceleration segment to the CDP and ends when the rotorcraft is brought to a complete stop on the designated takeoff surface. The critical engine is made inoperative at the CDP and the landing must be made with the remaining engine(s) operating within approved limits. The rejected takeoff distance is normally measured at a given reference point on the rotorcraft from the start of the takeoff to the same reference point after the rotorcraft has come to a complete stop. This distance should be increased by the rotorcraft length (including main and tail rotor tip paths).

(3) Takeoff Climbout Path.

(i) The "OEI transition segment" is defined as the segment from CDP where the engine becomes inoperative to V_{TOSS}. It is assumed that the maximum approved OEI power is used until the allowable time duration for that power is exhausted. It must be possible for the crew to fly the rotorcraft to V_{TOSS} and attain an altitude of 35 feet and then climb to 100 feet above the takeoff surface by flying the rotorcraft solely by the primary flight controls (including collective). The landing gear may be retracted after attaining a height of 35 feet above the takeoff surface, a speed of V_{TOSS}, and a positive rate of climb. Flight manual procedures may recommend adjustment of auxiliary controls to improve OEI performance. However, compliance with the performance requirements of § 29.67(a)(1) should not be based on use of secondary engine controls such as beepers, etc. Manipulation of the throttle controls or beep switches may be permitted for compliance with the performance requirements of § 29.67(a)(2) as long as such manipulation can be accomplished readily by the pilot flying the rotorcraft without removing his hands from the cyclic and collective flight controls. These manipulations of secondary engine controls should not make major adjustments in the power, and should not occur before attaining V_{TOSS}. There should be a minimum delay of 3 seconds after the critical engine is made inoperative before adjustment of secondary engine controls is allowed during the takeoff path determination. The failure of one engine cannot affect continued safe operation of the remaining engines or require any immediate action by the crew per § 29.903(b). If a 2 ½-minute power rating is used, it should be possible to complete the Category A takeoff profile (assuming an engine failure at CDP), accelerate to V_{TOSS}, attain 35 feet above the surface, and complete landing gear retraction prior to exhausting the 2 ½-minute time limit.

(ii) The takeoff safety speed, V_{TOSS}, is a speed at which 100 FPM rate of climb is assured under conditions defined in § 29.67(a)(1). The takeoff distance is the distance from initial hover to the point at which V_{TOSS} and 35 feet in a climbing posture are attained.

(4) Continued Climbout Path. Continued acceleration and climb capability from 100 feet above the takeoff surface is assured by the 100 FPM V_{TOSS} climb requirement of § 29.67(a)(1) and the 150 FPM requirement of § 29.67(a)(2), normally demonstrated at V_Y . It should be shown that the rotorcraft can be accelerated from V_{TOSS} to V_Y in a continuous maneuver without losing altitude, including any configurative change (landing gear retraction, etc.).

b. Procedures.

(1) Instrumentation. A photo theodolite, grid camera, or other position measuring equipment is required together with a ground station to measure wind, OAT, humidity (if applicable), and a two-way communication system to coordinate activities with the aircraft. A crash recovery team with support of a fire engine is highly desirable. Aircraft instrumentation should record with a time scale: engine parameters (speed, temperature, and power), rotor speed, flight parameters (airspeed, altitude, and normal acceleration as a minimum), flight control positions, power lever position, and landing gear loads. Additionally, a method should be devised to allow correlation of the aircraft instrumentation data with the space position data to accurately determine the length of the various takeoff segments.

(2) Establishing the Critical Decision Point (CDP).

(i) The CDP should be definable with the minimum crew using standard cockpit instrumentation. If a radar altimeter is used, it should be included in the minimum equipment list. If barometric altitude is used to define CDP, the operating conditions at which the altimeter is set should be defined. This is normally done on the ground with the minimum collective pitch. If the wind influences the altimeter reading, the correct relative wind information should be provided. Unless the rotorcraft is capable of hovering with one engine inoperative at the desired Category A weight, the CDP becomes largely a function of the surface area required for takeoff. If takeoff conditions scheduled include considerable surface area (on the order of 2,000 feet), the CDP airspeed may be a high value near V_Y. This will allow a higher takeoff weight and demonstrate compliance with the V_{TOSS} climb requirement of § 29.67(a)(1). In this case, the requirements of § 29.67(a)(2) usually become limiting. If required surface area is a small value, CDP will necessarily be some lower airspeed value to allow for an aborted takeoff on the available surface. Weight may need to be reduced at lower values of CDP airspeed (significantly below V_Y) to allow compliance with the climb requirement of § 29.67(a)(1). Compliance with climb requirements can be substantiated initially by testing at a safe altitude above the ground. When OEI climb conditions are verified for weight, configuration, pressure altitude, and temperature, the CDP is then evaluated in a rejected takeoff.

(ii) A Category A takeoff procedure for which the CDP is defined as a specific "time," "height," or "airspeed" in the takeoff sequence combined with a precise takeoff crew procedure may be approved on the basis of equivalent safety when the following conditions can be satisfied:

(A) The flightcrew takeoff procedure must be shown to be consistently repeatable and not require exceptional piloting skill.

(B) It must be documented that the takeoff procedure will produce the required minimum energy level in terms of height and airspeed for all combinations of gross weight, altitude, and ambient temperature for which takeoff data are scheduled. This may best be accomplished by conducting takeoff procedure abuse tests to show that variations from the established takeoff procedure that could reasonably be expected to occur in service do not result in significant increases in the takeoff distances.

(3) Rejected Takeoff Distance. The rejected takeoff is similar in many respects to the height-velocity (HV) tests described in paragraph AC 29.73. Most of the comments, cautions, and techniques for HV also apply here even though typical flight conditions at CDP are less critical than limiting HV points. As mentioned in paragraph AC 29.79, a minimum 5-knot clearance from any HV limiting condition should be provided throughout the takeoff flight path (see figure AC 29.63-1), and tests should be conducted simulating an unplanned engine cut. The HV diagram appropriate in the Category A test weights may be much less restrictive than that determined for Category B conditions. Normally, a minimum 1-second delay is applied after engine failure before pilot collective control corrections are allowed. However, if pilot cues are strong enough to make engine failure unmistakable, normal pilot reaction time may be utilized following engine failure. As in all engine failure testing, the pilot should not anticipate the failure by changing flight control positions or aircraft attitude. Average pilot techniques should be used. The two primary objectives of rejected takeoff testing are an assured capability to safely return to the takeoff surface when an engine fails at any point prior to CDP and the determination of the rejected takeoff distance that is needed when an engine fails at the CDP. It is important that the surface conditions be defined. For the rejected takeoff distance tests, a minimum of five satisfactory runs should be flown by the FAA/AUTHORITY pilot. The rejected takeoff distances from company and FAA/AUTHORITY runs may be averaged. The rejected takeoff distance tests will be used together with the OEI continued takeoff profiles to establish the required surface area for Category A operations.

(4) Continued Takeoff Distance.

(i) Continued takeoff profiles should be flown to determine the continued takeoff distance. This distance is measured from the point of takeoff initiation to the point in the takeoff profile where the following three conditions have all been attained after a failure of the critical engine at CDP: an airspeed equal to or greater than V_{TOSS}, a positive rate of climb, and a height of at least 35 feet above the takeoff surface. The rotorcraft should not contact the ground at any point after engine failure. If the rotorcraft descends below 35 feet above the takeoff surface while accelerating to V_{TOSS}, the takeoff distance is extended to the point that 35 feet is reattained with a positive rate of climb.

(ii) If the CDP is significantly above 35 feet so that the rotorcraft does not descend below 35 feet during acceleration to V_{TOSS}, the takeoff distance then becomes the distance to the point in the takeoff profile at which both V_{TOSS} and a positive rate-of-climb are attained after failure of the critical engine at CDP. For most applications, the rotorcraft should not be allowed to descend more than one-half the CDP height above the takeoff surface while accelerating to V_{TOSS}. In addition, the rotorcraft should not be allowed to descend below the height above the takeoff surface at which a landing flare would normally be initiated. For example, if a rotorcraft has a CDP of 20 feet but when landing would normally initiate the landing flare at 15 feet, the takeoff profile should not be allowed to descend to 10 feet but should remain above 15 feet in establishing the takeoff distances.

(iii) In establishing the continued takeoff distance, the applicable pilot recognition delay time should be applied following the engine failure at CDP, and the takeoff profile should be established with the pilot using primary flight controls only to control the rotorcraft. The pilot engine failure recognition time delay before adjustment of the collective pitch control should be a minimum of 1 second unless it can be demonstrated that the pilot will have unmistakable engine failure cues sooner than 1 second.

(iv) Engine failure testing should be initially conducted at a safe distance above the ground to assess the continued takeoff profile before conducting the actual profiles for credit. This procedure will serve to validate predicted performance and may prevent an unexpected return to the surface during continued takeoff tests. A minimum of five acceptable runs should be flown by the FAA/AUTHORITY pilot, and these should be averaged with five acceptable runs flown by the manufacturer’s pilot.

(5) <u>Abuse Testing</u>. Takeoff procedure abuse tests should be conducted to show that reasonably expected variations in service from the established takeoff procedures do not result in a significant increase in the established takeoff distances. Variations should include such considerations as under or over rotation during the takeoff initiation, under or over application of acceleration power, and missed CDP target parameters (e.g., time, height, or airspeed).

(6) <u>Continued Climbout Path</u>. The climb performance requirements of § 29.67(a)(1) should be met at the end of the continued takeoff distance segment. Beginning at this point, the landing gear may be retracted, and secondary engine controls may be manipulated to adjust power. Any manipulation of secondary engine

controls should be accomplished readily by the pilot flying the rotorcraft without removing his hands from the cyclic and collective flight controls. The climb should be continued at V_{TOSS} until approximately 100 feet above the takeoff surface. It should be demonstrated that the rotorcraft including any configuration changes can be accelerated from V_{TOSS} to V_Y in a continuous maneuver without losing altitude. The airspeed and rotorcraft configuration (landing gear position, rotor RPM engine power, etc.) used to show compliance with the climb requirements of § 29.67(a)(2) should be attained at or prior to reaching 1,000 feet above the takeoff surface.

(7) Power. Power should be limited to minimum specification values on the operating engine(s). This may be accomplished by adjustment of the engine topping to minimum specification values including consideration of temperature effects on engine power. Turbine engine power does not vary directly with density altitude (H_D). At a given H_D, turbine engine power available varies with ambient temperature. Turbine engines typically produce less horsepower as ambient temperature is increased (pressure altitude decreases) at a given density altitude, although some engines produce less horsepower at extremely cold temperatures. In either event, if one test sequence is to be utilized for a given H_D, it would be appropriate to restrict test power to the lowest value attainable from a minimum specification engine through the approved ambient temperature range at the density altitude of the test. To attain maximum weights for varying ambient conditions, the applicant may utilize a parametric mapping of power available, pressure altitude, and temperature effects. For this case, engine topping may be adjusted throughout a range appropriate to the test H_D.

(8) Aircraft Loading. Both forward and aft CG extremes should be spot checked to determine the critical loading for takeoff distances. Forward center of gravity is usually critical for continued takeoff distance tests while aft CG may be critical for the rejected takeoff because of over-the-nose visibility. A minimum of two weights should be flown at each altitude if the manufacturer elects to schedule field length variation as a function of gross weight. One weight should be the maximum weight for prevailing conditions and the other weight(s) should be low enough to attain a sufficient spread to verify weight accountability.

(9) Extrapolation. Weight cannot be extrapolated above test weight for the same reasons discussed in paragraph AC 29.79. See paragraph AC 29.45 regarding altitude extrapolation of test results.

(10) Ambient Conditions. Appropriate test limits for ambient conditions such as wind and temperature are contained in paragraph AC 29.45. Test data must be corrected for existing wind conditions during takeoff distance testing. Credit for headwind conditions may be given during flight manual data expansion. Refer to paragraph AC 29.45(b)(1) under "Winds for Testing" for allowable wind credit. Care should be applied in considering headwind credit for vertical operations as previous experience has resulted in difficulty collecting meaningful, repeatable data.

(11) Vertical Takeoffs.

(i) General. Guidelines for rotorcraft certification using vertical takeoff techniques were developed and utilized for civil certification programs many years ago. As experience has been gained, certain policy decisions have modified these guidelines. The following guidelines incorporate all available policy information as of January 1, 1981. The reader should be familiar with the preceding discussion regarding conventional Category A takeoff profiles because duplicate information is not repeated here.

(ii) Takeoff Profile. A typical vertical takeoff profile for a ground level heliport is shown in figure AC 29.59-1. The maneuver begins with the addition of sufficient power to initiate a climb to the CDP. It must be possible to make a safe landing without exceptional pilot skill if an engine fails at any point up to the CDP. At the CDP, the pilot becomes “go” oriented and continues the takeoff if an engine fails. A typical profile for pinnacle takeoff conditions is shown in figure AC 29.59-2. Considerations are similar to those of the ground level heliport in figure AC 29.59-1; however, the OEI pinnacle profile allows descent below the takeoff surface, specifies minimum edge clearance criteria, and allows relaxed requirements for final segment climb. Thus far, descent profiles up to 50 feet below the takeoff surface have been allowed; however, there is no reason why greater values could not be determined during engineering flight tests for certification. Use of such a profile, of course, would be dependent on obtaining an operational approval.

(iii) Critical Decision Point (CDP). For vertical takeoffs, the climb to CDP is nearly vertical, and CDP is typically defined primarily by height. Sufficient testing must be conducted to define a band of CDP conditions (heights) which will be consistent with anticipated variations in pilot technique and the minimum amount of equipment to be installed on the production aircraft. Rejected takeoffs are most critical from high CDPs, and continued OEI takeoffs are most critical from low heights. Tests at the extremes of this band are intended to verify that the anticipated CDP band is safe and repeatable in service for reasonable variations in pilot technique. These extreme points should not be used for distance determination when averaging takeoff performance data.

(iv) Conduct of the Test. Vertical takeoff profiles must be flown from a pad simulating operational conditions because the sight picture may be critical to successful OEI operations, particularly for elevated heliports. At all points on the vertical takeoff flight path up to the CDP, the pilot, with reasonable head movement, shall be able to keep sufficient portions of two heliport boundaries (front and one side) or equivalent markings in view to achieve a safe landing in case of engine failure. Normally, a minimum 1-second delay is applied after engine failure before pilot collective control corrections are allowed. However, if pilot cues are strong enough to make engine failure unmistakable, normal pilot reaction time may be utilized following engine failure.

(A) Establish the rejected takeoff distance as the horizontal distance from the rearmost point of the rotorcraft at the initiation of takeoff to the foremost point after the rotorcraft comes to a stop on the takeoff surface (including rotor tip path), assuming an engine failure in the vertical climb at the CDP; or

(B) Establish the continued takeoff distance as the horizontal distance from lift-off to the point at which, following engine failure at CDP, the rotorcraft achieves 35 feet above the takeoff surface and V_{TOSS} in a climbing posture. The continued takeoff profile from elevated heliports must clear the heliport obstructions by at least 15 feet vertically and 35 feet horizontally.

(v) Climb Requirements.

(A) The OEI takeoff profile should include a climb at V_{TOSS} to 200 feet above the takeoff surface prior to accelerating to a higher speed.

(B) For elevated heliports, the climb requirement of § 29.67(a)(2) may be met at 200 feet above the takeoff surface or 1,000 feet above the surrounding terrain, whichever is higher.

(vi) Extrapolation. Basic guidelines for extrapolation are contained in paragraph AC 29.45. If, however, vertical takeoff weights are based upon allowable weights for hovering out-of-ground effect (OGE) with one engine inoperative, all vertical takeoff performance aspects may be extrapolated to the highest altitude requested for takeoff and landing.

(12) Night Operations.

(i) A minimum of three normal takeoffs (and landings) should be conducted to assure that aircraft lighting (internal and external) is adequate to allow normal Category A operations at night.

(ii) Engine failures should be simulated from points along the recommended takeoff profile. Night OEI rejected takeoffs and continued takeoffs from the CDP should be conducted to assure adequate night field of view and realization of Category A field lengths.

(iii) If special airfield markings are used as a reference or to define the CDP, the aircraft external lighting should be evaluated to assure that these airfield markings are adequately visible for night operations.

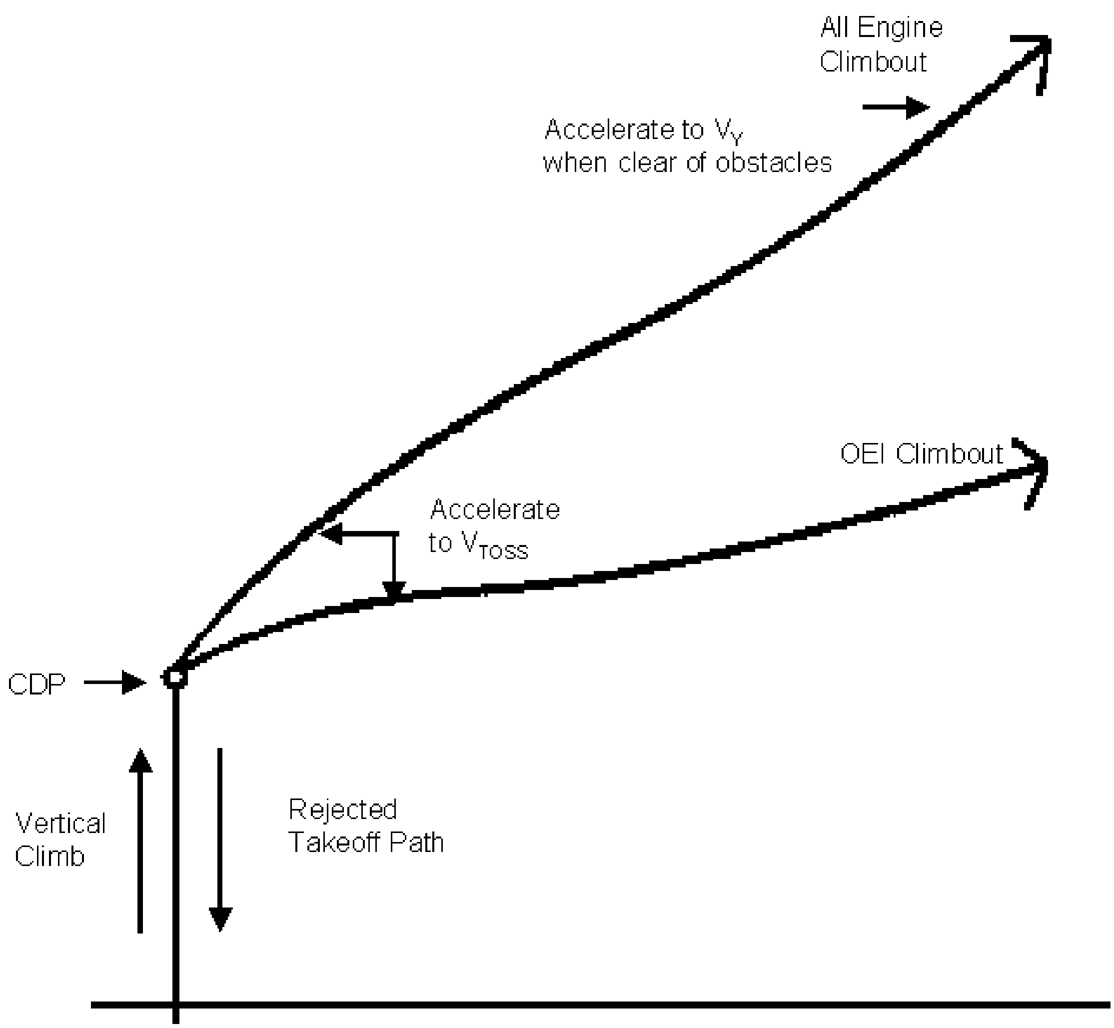


FIGURE AC 29.59-1 CATEGORY A VERTICAL TAKEOFF PROFILE
GROUND LEVEL HELIPORT

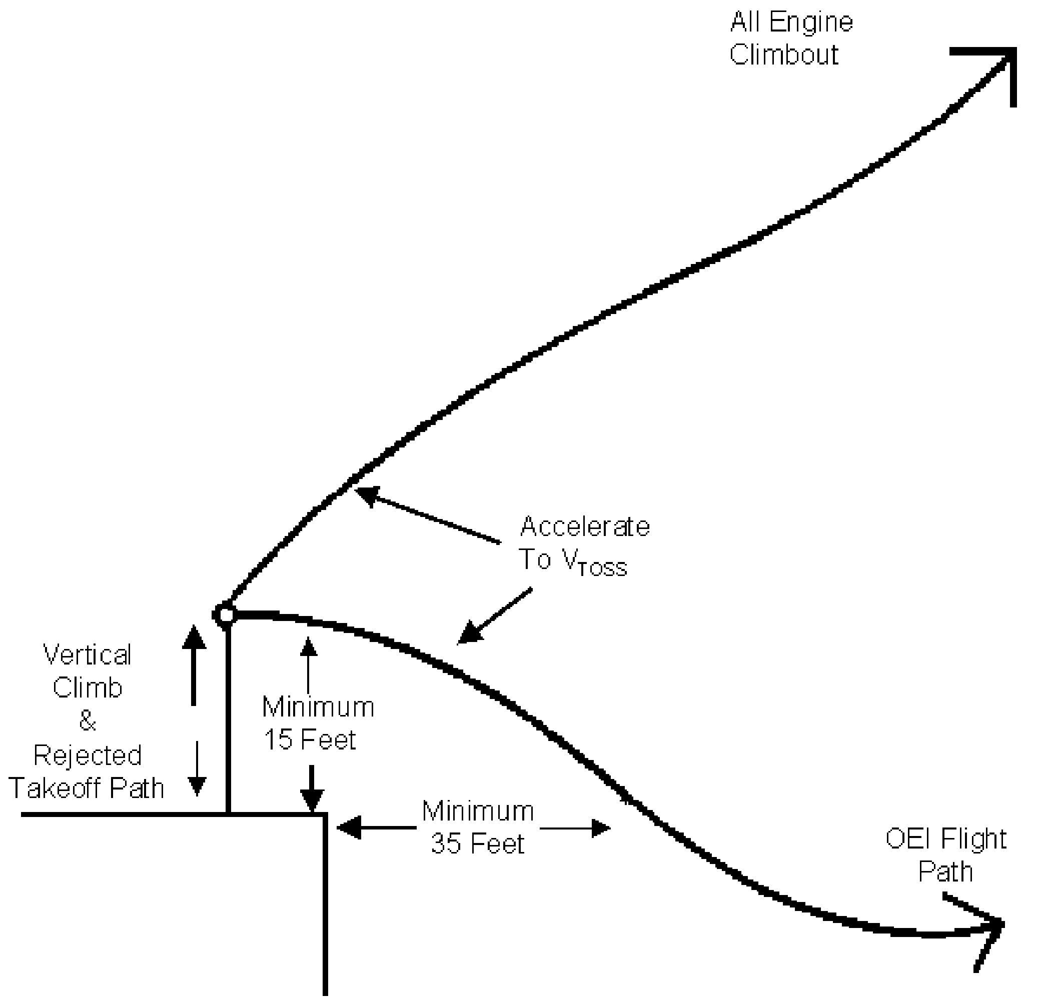


FIGURE AC 29.59-2 CATEGORY A VERTICAL TAKEOFF PROFILE PINNACLE

AC 29.59A (AC's 29.60, 29.61, & 29.62) §§ 29.59 (29.60, 29.61 and 29.62) (Amendment 29-39) TAKEOFF PATH, DISTANCE AND REJECTED TAKEOFF; GROUND LEVEL AND ELEVATED HELIPORT: CATEGORY A

(For § 29.59 prior to Amendment 39, see paragraph AC 29.59.)

a. Explanation. Amendment 29-39 moved the rejected takeoff requirements from § 29.55 to a new § 29.62 and clearly defined the takeoff path. It also added new §§ 29.60 and 29.61 to introduce the requirements for elevated heliport takeoff path, Category A and to more clearly define the parameters to be used in determining takeoff distance, respectively.

(1) Takeoff Decision Point. The Category A concept limits the rotorcraft takeoff weight such that if an engine failure is recognized at or before the TDP, a safe landing can be made or if an engine failure is recognized at or after the TDP, the takeoff can be continued. The purpose of these tests is to define the TDP, evaluate the necessary pilot techniques, and determine the required takeoff area for either alternative. The condition of equal distances for either stopping or continuing the takeoff is called a "balanced" field length. The combination of altitude and speed at the TDP which produces a balanced field length is not required for certification. This section deals with the Category A takeoff and rejected takeoff profiles. The profiles necessarily involve consideration of an average pilot skill level as well as a sequence in which it is assumed various configuration adjustments are made to the rotorcraft.

(2) Takeoff. The Category A takeoff path begins with an all-engines-operating acceleration segment to the engine failure point and continues with a one-engine-inoperative acceleration through the TDP to the takeoff safety speed (V_{TOSS}). The engine failure point (EFP) and TDP are separated by pilot recognition time. (See Conventional Takeoff Profile, figure AC 29.53A-1, paragraph AC 29.53A of this advisory circular.) TDP is a "go/no-go condition which is analogous to V1 speed in transport airplanes. Prior to TDP the pilot is "stop" oriented, and when an engine failure is recognized in this portion of the takeoff, the pilot will abort because the rotorcraft has not yet achieved sufficient energy to assure continued flight. At the TDP the pilot becomes "go" oriented and when an engine failure is recognized at or beyond this point, the pilot will continue the takeoff because sufficient surface area no longer remains for an aborted takeoff. The takeoff flight path and the TDP should be defined such that a safe landing can be made from any point up to the TDP. This profile may differ significantly from the takeoff flight path developed for Category B weights. The TDP is the last point in the takeoff profile at which a rejected takeoff capability within the scheduled takeoff surface distance is assured. If an engine failure does not occur, the pilot continues the climb and accelerates past the TDP to the recommended climb speed.

(3) Rejected Takeoff. The rejected takeoff profile begins with an all engine acceleration segment to the EFP and ends when the rotorcraft is brought to a complete stop on the designated takeoff surface. The critical engine is made inoperative prior to

the TDP, and the landing should be made with the remaining engine(s) operating within approved limits. The rejected takeoff distance is normally measured at a given reference point on the rotorcraft from the start of the takeoff to the same reference point after the rotorcraft has come to a complete stop. This distance should be increased by the rotorcraft length (including main and tail rotor tip paths).

(4) Takeoff Path.

(i) The transition to OEI flight takes place between the engine failure point and the point at which V_{TOSS} is achieved. It is assumed that the maximum approved OEI power is used until the allowable time duration for that power is exhausted. It should be possible for the crew to fly the rotorcraft to V_{TOSS} and attain an altitude of 35 feet and positive rate of climb and then climb to 200 feet above the takeoff surface or the lowest point in the takeoff path by flying the rotorcraft solely by the primary flight controls (including collective). At no time during the takeoff shall the rotorcraft descend below 15 feet above the takeoff surface when the TDP is above 15 feet. The landing gear may be retracted after attaining a speed of V_{TOSS}, and a positive rate of climb. Flight manual procedures may recommend adjustment of auxiliary controls to improve OEI performance, but compliance with the performance requirements of § 29.67(a)(1) may not be based on use of secondary engine controls such as RPM beep switches. During the continued takeoff profile, the pilot is assumed to be flying the rotorcraft via the primary flight controls (cyclic stick, collective, and directional pedals). Manipulation of the throttle controls or beep switches may be permitted as long as such manipulation can be accomplished readily by the pilot flying the rotorcraft without removing his hands from the cyclic and collective flight controls. These manipulations of engine controls should not make major adjustments in power and should not occur before attaining V_{TOSS}. In no case should this be less than 3 seconds after the critical engine is made inoperative. The failure of one engine cannot affect continued safe operation of the remaining engines or require any immediate action by the crew per § 29.903(b). If a 30-second/2-minute or a 2 ½-minute power rating is used, it should be possible to complete the Category A takeoff profile (assuming recognition of an engine failure at or prior to the TDP), accelerate to V_{TOSS}, attain 35 feet above the surface, stabilize in a climb of at least 100 feet per minute, and complete landing gear retraction prior to exhausting the 2 ½-minute time limit.

(ii) The takeoff safety speed, V_{TOSS}, is a speed at which 100 FPM rate of climb is assured under conditions defined in § 29.67(a)(1). The takeoff distance is the distance from the start of the takeoff to the point at which V_{TOSS}, 35 feet above the takeoff surface, and a positive rate of climb are attained.

(5) Continued Climbout Path. Continued acceleration and climb capability are assured by the 100 FPM V_{TOSS} climb requirement of § 29.67(a)(1) and the 150 FPM requirement of § 29.67(a)(2), normally demonstrated at V_y. It should be shown that the rotorcraft can be accelerated from V_{TOSS} to V_Y in a continuous maneuver without losing altitude, including any configurative change (landing gear retraction, etc.). The distance

required to accelerate from V_{TOSS} to V_Y must be considered in determination of the climb and gradients required by § 29.1587(a)(6)(i) and (a)(6)(ii).

b. Procedures.

(1) Instrumentation. A photo theodolite, grid camera, GPS, or other position measuring equipment is normally required together with a ground station to measure wind, OAT, humidity (if applicable), and a two-way communication system to coordinate activities with the aircraft. A crash recovery team with support of a fire engine is highly desirable. Aircraft instrumentation should record with a time scale: engine parameters (speed, temperature, and power), rotor speed, flight parameters (airspeed, altitude, and normal acceleration as a minimum), flight control positions, power lever position, and landing gear loads. Additionally, a method should be devised to allow correlation of the aircraft instrumentation data with the space position data to accurately determine the length of the various takeoff segments.

(2) Establishing the Takeoff Decision Point (TDP).

(i) The TDP should be definable with the minimum crew using standard cockpit instrumentation. If a radar altimeter is used, it should be included in the minimum equipment list. If barometric altitude is used to define TDP, the operating conditions at which the altimeter is set should be defined. This is normally done on the ground with the minimum collective pitch. If the wind influences the altimeter reading, the correct relative wind information should be provided. Unless the rotorcraft is capable of hovering with one engine inoperative at the desired Category A weight, the TDP becomes largely a function of the surface area required for takeoff. If takeoff conditions scheduled include considerable surface area (on the order of 2,000 feet), the TDP airspeed may be a high value near V_Y. This will allow a higher takeoff weight and demonstrate compliance with the V_{TOSS} climb requirement of § 29.67(a)(1). In this case, the requirements of § 29.67(a)(2) usually become limiting. If required surface area is a small value, TDP will necessarily be some lower airspeed value to allow for an aborted takeoff on the available surface. Weight may need to be reduced at lower values of TDP airspeed (significantly below V_Y) to allow compliance with the climb requirement of § 29.67(a)(1). Compliance with climb requirements can be substantiated initially by testing at a safe altitude above the ground. When OEI climb conditions are verified for weight, configuration, pressure altitude, and temperature, the TDP is then evaluated in a rejected takeoff.

(ii) A Category A takeoff procedure should satisfy the following conditions:

(A) The flightcrew takeoff procedure should be shown to be consistently repeatable and not require exceptional piloting skill.

(B) It should be documented that the takeoff procedure will produce the required minimum energy level in terms of height and airspeed for all combinations of gross weight, altitude, and ambient temperature for which takeoff data are scheduled. This may best be accomplished by conducting takeoff procedure abuse tests to show that variations from the established takeoff procedure that could reasonably be expected to occur in service do not result in significant increases in the takeoff distances.

(3) Rejected Takeoff Distance. The rejected takeoff is similar in many respects to the height-velocity (HV) tests described in paragraph AC 29.73. Most of the comments, cautions, and techniques for HV also apply here even though typical flight conditions at TDP are less critical than limiting HV points. As mentioned in paragraph AC 29.79, a minimum 5-knot clearance from any HV limiting condition should be provided throughout the takeoff flight path (see figure AC 29.63-1), and tests should be conducted simulating an unplanned engine cut. The HV diagram appropriate to the Category A test weights may be much less restrictive than that determined for Category B conditions. Normally, a minimum 1-second delay (or pilot reaction time, whichever is greater) is applied after engine failure recognition, before pilot collective control corrections are allowed. If the rotorcraft incorporates an engine failure warning device, engine failure recognition should not be less than the time required for the engine to spool down and activate the device. As in all engine failure testing, the pilot should not anticipate the failure by changing flight control positions or aircraft attitude. Average pilot techniques should be used. The two primary objectives of rejected takeoff testing are an assured capability to safely return to the takeoff surface when an engine failure is recognized at any point prior to TDP and the determination of the rejected takeoff distance required. It is important that the surface conditions be defined. The rejected takeoff distance tests will be used together with the OEI continued takeoff profiles to establish the required surface area for Category A operations.

(4) Takeoff Distance.

(i) Continued takeoff profiles should be flown to determine the continued takeoff distance. This distance is measured from the point of takeoff initiation to the point in the takeoff profile where the following three conditions have all been attained after a failure of the critical engine prior to TDP: an airspeed equal to or greater than V_{TOSS}, a positive rate of climb, and a height of at least 35 feet above the takeoff surface. If the rotorcraft descends below 35 feet above the takeoff surface while accelerating to V_{TOSS}, the takeoff distance is extended to the point that 35 feet is reattained with a positive rate of climb.

(ii) If the TDP is significantly above 35 feet so that the rotorcraft does not descend below 35 feet during acceleration to V_{TOSS}, the takeoff distance then becomes the distance to the point in the takeoff profile at which both V_{TOSS} and a positive rate of climb are attained after failure of the critical engine prior to the TDP. For all applications, rotorcraft should not be allowed to descend below 15 feet above the takeoff surface while accelerating to V_{TOSS} when TDP is above 15 feet. When TDP is below 15 feet, the aircraft should be able to accelerate in level flight or climb. Fifteen feet should be considered the absolute minimum clearance allowed with greater clearances required for some rotorcraft dependent on rotorcraft geometry and performance characteristics. In addition, the rotorcraft should not be allowed to descend below the height above the takeoff surface at which a landing flare would normally be initiated. For example, a medium size twin-engined rotorcraft with a TDP of 100 feet or greater, using 20° nose down, would be expected to clear the ground by 25 feet whereas a large multiengined rotorcraft, using similar attitudes and TDP's, would be expected to clear by 35 feet. For elevated heliports the rotorcraft may descend below the landing surface, but all parts of the rotorcraft must clear the heliport and all other obstacles by not less than 15 feet. These minimum heights would need to be demonstrated with variations in piloting techniques and with pilot recognition and reaction times for engine failures occurring before and after TDP.

(iii) In establishing the continued takeoff distance, the applicable pilot recognition delay time should be applied following the engine failure prior to the TDP, and the takeoff profile should be established with the pilot using primary flight controls only to control the rotorcraft. The pilot engine failure recognition time delay before adjustment of the collective pitch control should be a minimum of 1 second.

(iv) Engine failure testing should be initially conducted at a safe distance above the ground to assess the continued takeoff profile before conducting the actual profiles for credit. This procedure will serve to validate predicted performance and may prevent an unexpected return to the surface during continued takeoff tests. A minimum of five acceptable runs should be flown by the FAA/AUTHORITY pilot, and these should be averaged with five acceptable runs flown by the manufacturer's pilot.

(5) Abuse Testing. Takeoff procedure abuse tests should be conducted to show that reasonably expected variations in service from the established takeoff procedures do not result in a significant increase in the established takeoff distances. Variations should include such considerations as under or over rotation during the takeoff initiation, under or over application of acceleration power, and missed TDP target parameters (e.g., time, height, or airspeed).

(6) Continued Climbout Path. The landing gear may be retracted at 35 feet. The climb should be continued at V_{TOSS} until 200 feet above the takeoff surface. The climb requirements of § 29.67(a)(1) should be met at 200 feet. It should be demonstrated that the rotorcraft, including any configuration changes, can be accelerated from V_{TOSS} to V_Y in a continuous maneuver without losing altitude. The airspeed and rotorcraft configuration (landing gear position, rotor RPM engine power, etc.) used to show compliance with the climb requirements of § 29.67(a)(2) should be attained at or prior to reaching 1,000 feet above the takeoff surface.

(7) Power. Power should be limited to minimum specification values on the operating engine(s). This may be accomplished by adjustment of the engine topping to minimum specification values including consideration of temperature effects on engine power. Turbine engine power does not vary directly with density altitude (H_D). At a given H_D, turbine engine power available varies with ambient temperature. Turbine

engines typically produce less horsepower as ambient temperature is increased (pressure altitude decreases) at a given density altitude, although some engines produce less horsepower at extremely cold temperatures. In either event, if one test sequence is to be utilized for a given H_D, it would be appropriate to restrict test power to the lowest value attainable from a minimum specification engine through the approved ambient temperature range at the density altitude of the test. To attain maximum weights for varying ambient conditions, the applicant may utilize a parametric mapping of power available, pressure altitude, and temperature effects. For this case, engine topping may be adjusted throughout a range appropriate to the test H_D.

(8) Aircraft Loading. Both forward and aft CG extremes should be spot checked to determine the critical loading for takeoff distances. Forward center of gravity is usually critical for continued takeoff distance tests while aft CG may be critical for the rejected takeoff due to forward/downward field of view. A minimum of two weights should be flown at each altitude if the manufacturer elects to schedule field length variation as a function of gross weight. One weight should be the maximum weight for prevailing conditions and the other weight(s) should be low enough to attain a sufficient spread to verify weight accountability.

(9) Extrapolation. Takeoff and landing data may be extrapolated up to 4000 feet along an established W/σ line, to the maximum gross weight of the rotorcraft. However, extrapolation will not be considered valid if unacceptable or marginally acceptable landing gear loads are experienced during testing at weights below the W/σ limit. See paragraph AC 29.77b(5) for further discussion of landing gear loads.

(10) Ambient Conditions. Appropriate test limits for ambient conditions such as wind and temperature are contained in paragraph AC 29.45. Test data should be corrected for existing wind conditions during takeoff distance testing. Credit for headwind conditions may be given during flight manual data expansion. Refer to paragraph AC 29.1587(a)(3)(iii) under “Wind Accountability” for allowable wind credit. Care should be applied in considering headwind credit for vertical operations as previous experience has resulted in difficulty collecting meaningful, repeatable data.

(11) Vertical Takeoffs.

(i) General. Guidelines for rotorcraft certification using vertical takeoff techniques were developed and utilized for civil certification programs many years ago. As experience has been gained, certain policy decisions have modified these guidelines. The reader should be familiar with the preceding discussion regarding conventional Category A takeoff profiles because duplicate information is not repeated here.

(ii) Takeoff Profile. A typical vertical takeoff profile for a ground level heliport is shown in figure AC 29.59A-1. The maneuver begins with the addition of sufficient power to initiate a climb to the TDP. It should be possible to make a safe landing without exceptional pilot skill if an engine fails at any point up to the TDP less engine failure recognition time. At the TDP, the pilot becomes "go" oriented and continues the takeoff if an engine fails. The rotorcraft should not be allowed to descend below 15 feet above the takeoff surface during the continued takeoff. A typical profile for elevated heliports takeoff conditions is shown in figure AC 29.59A-2. Descent profile below the takeoff surface is allowed, after clearing the platform by at least a 15 feet radial margin, provided that the drop down height from the takeoff surface and the distance to reach V_{TOSS} with a positive rate of climb is given in the performance chapter of the RFM.

(iii) Takeoff Decision Point (TDP). For vertical takeoffs, the climb to the TDP is nearly vertical, and the TDP is typically defined primarily by height. Sufficient testing should be conducted to define a band of TDP conditions (heights) which will be consistent with anticipated variations in pilot technique and the minimum amount of equipment to be installed on the production aircraft. Rejected takeoffs are most critical from high TDP's, and continued OEI takeoffs are most critical from low heights. Tests at the extremes of this band are intended to verify that the anticipated TDP band is safe and repeatable in service for reasonable variations in pilot technique. These extreme points should not be used for distance determination when averaging takeoff performance data.

(iv) Conduct of the Test. Vertical takeoff profiles should be flown from a pad simulating operational conditions because the sight picture may be critical to successful OEI operations, particularly for elevated heliports. At all points on the vertical takeoff flight path up to the TDP, the pilot, with reasonable head movement, shall be able to keep sufficient portions of two heliport boundaries (front and one side) or equivalent markings in view to achieve a safe landing in case of engine failure. Normally, a minimum 1-second delay or pilot recognition time interval, whichever is greater, is applied after the EFP before pilot collective control corrections are allowed. If the rotorcraft incorporates an engine failure warning device, engine failure recognition should not be less than the time required for the engine to spool down and activate the device.

(A) Establish the rejected takeoff distance as the horizontal distance from the rearmost point of the rotorcraft at the initiation of takeoff to the foremost point after the rotorcraft comes to a stop on the takeoff surface (including rotor tip path), assuming an engine failure in the vertical climb at the TDP.

(B) Establish the continued takeoff distance as the horizontal distance from lift-off to the point at which, following engine failure prior to the TDP, the rotorcraft achieves; for a ground level heliport, 35 feet above the takeoff surface and V_{TOSS} with a positive rate of climb; for an elevated heliport, the lowest point of the takeoff profile and not less than V_{TOSS} with a positive rate of climb. The continued takeoff profile from elevated heliports should clear the heliport obstructions by at least a 15 feet radial margin.

(C) When used, the back-up technique usually requires the pilot to keep sufficient portions of the helipad in view and involves a rearward movement from the takeoff point to the TDP. In such cases the rearward horizontal distance required should be established as the distance from the rearmost point of the rotorcraft at the initiation of takeoff to the rearmost part of the rotorcraft at TDP.

(D) If special helipad markings or other non-standard external references are required to achieve the vertical takeoff performance, these special references should be included in the limitations section of the RFM.

(v) Climb Requirements.

(A) Ground level heliport. The OEI takeoff profile should include a climb at V_{TOSS} to 200 feet above the takeoff surface then an acceleration in level flight from V_{TOSS} to V_Y and a climb at V_Y to 1000 feet above the lowest point of the takeoff profile. The climb requirements of § 29.67(a)(1) and (a)(2) may be met at referenced points located respectively at 200 feet and 1000 feet above the takeoff surface. The distance required to accelerate from V_{TOSS} to V_Y must be considered in determination of the climb gradient required by § 29.1587 (a)(6)(i) and (a)(6)(ii).

(B) Elevated heliport. The OEI takeoff profile should include a climb at V_{TOSS} to 200 feet above the lowest point of the takeoff profile then an acceleration in level flight from V_{TOSS} to V_Y and a climb at V_Y to 1000 feet above the lowest point of the takeoff profile. The climb requirements of § 29.67(a)(1) and (a)(2) may be met at referenced points located respectively at 200 feet and 1000 feet above the lowest point of the takeoff profile.

(vi) Extrapolation. Basic guidelines for extrapolation are contained in paragraph AC 29.45. Weight can not be extrapolated above test weight. Altitude extrapolation should be limited to a maximum of $\pm$ 4000 feet.

(12) Night Operations.

(i) A minimum of three normal takeoffs (and landings) should be conducted to ensure that aircraft lighting (internal and external) is adequate to allow normal Category A operations at night.

(ii) Engine failures should be simulated from points along the recommended takeoff profile. Night OEI rejected takeoffs and continued takeoffs from the TDP should be conducted to ensure adequate night field of view and realization of Category A field lengths.

(iii) If special airfield marking/lighting is used as a reference or to define the TDP, the aircraft external lighting should be evaluated to ensure the airfield marking/lighting is adequately visible for night operations.

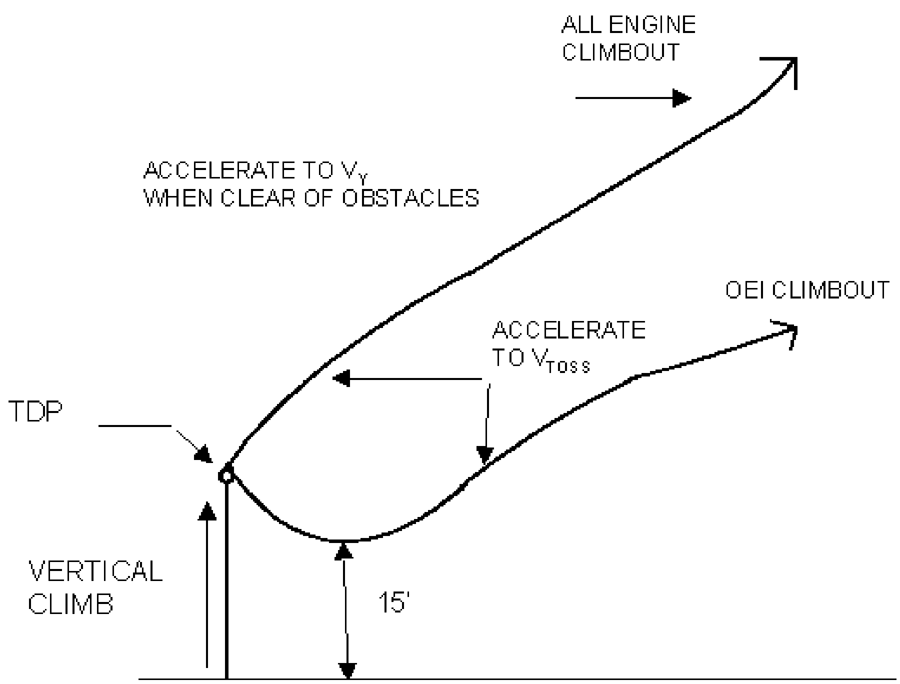


FIGURE AC 29.59A-1 CATEGORY A VERTICAL TAKEOFF PROFILE
GROUND LEVEL HELIPORT

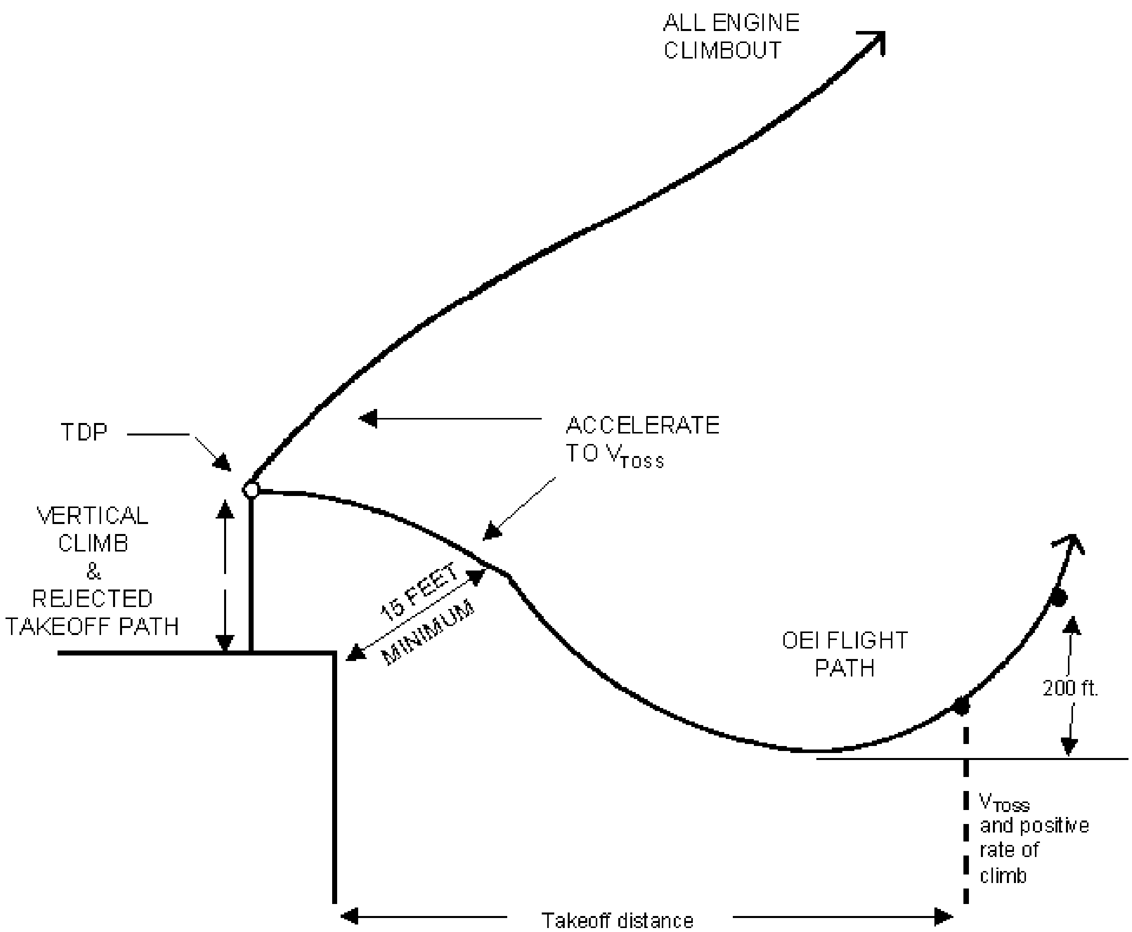


FIGURE AC 29.59A-2 CATEGORY A VERTICAL TAKEOFF PROFILE ELEVATED HELIPORT

AC 29.63. <u>§ 29.63 (Amendment. 29-12) TAKEOFF: CATEGORY B</u>.

a. <u>Explanation</u>.

(1) Takeoff distance is the horizontal distance measured from an initial position to a point 50 feet above the takeoff surface with all engines operating within approved limits.

(2) The height-velocity diagram is normally developed and accepted prior to conducting takeoff distance tests. Takeoff distance tests are conducted avoiding the critical areas of the diagram. The amount of power utilized in determining takeoff distance may not be greater than that used in constructing the takeoff corridor and "knee" portions of the height-velocity diagram. Power might also have to be constrained, depending upon the amount of excess power available, so that a "reasonable" nose down pitch attitude is not exceeded during the initial portion of the takeoff run. Acceptable values used during past programs include:

(i) Hover power + 10 percent (not to exceed rated engine takeoff power limits)

(ii) A percent transmission limiting torque (not to exceed rated engine takeoff power limits), and

(iii) Engine (or transmission) limiting power for the particular ambient conditions.

(3) The critical center of gravity should be used for takeoff distance tests. Critical center of gravity should be established analytically or from previous testing and may be forward or aft depending on the type of rotorcraft. Items that should be considered in determining the critical center of gravity are climb performance and cockpit visibility. At least two gross weights should be flown at each test altitude, if weight accountability is desired, in order to validate the manufacturers prediction of weight effects.

(4) The speed utilized at the 50-foot point in the takeoff profile (V_{50} speed) may be largely determined by the ability to obtain reliable, repeatable airspeed indications which can also comply with § 29.1323. Section 29.1323 ties the airspeed system accuracy requirements to the climbout speed. The climbout speed should be that speed attained at 50 feet in complying with § 29.63.

b. <u>Procedures</u>.

(1) <u>Instrumentation</u>. A ground station will measure ambient temperature, humidity (if applicable), and wind. For allowable wind conditions and engine power considerations refer to paragraph AC 29.45. A photo panel or hand recording method

may be utilized, as necessary, to record engine and flight parameters. A phototheodolite, takeoff and landing camera, or other approved instrumentation is utilized to measure distance, heights, speed, and time.

(2) Conduct of the Test. If the applicant elects to show weight effects on distance, at least two weights should be flown and, depending on the range of takeoff and landing altitudes to be approved, at least two test altitudes should be flown. Altitudes should be sufficiently far apart to include a major portion of the approved takeoff and landing altitude range. Takeoff profiles should be started from an initial condition. For takeoffs from a hover, the hover height should be determined by performing fixed collective takeoffs as described in paragraph AC 29.51. “Takeoff” power should be smoothly applied and the aircraft nose lowered as necessary to accelerate without gaining excessive altitude. It must be possible to conduct a consistent takeoff profile clear of the height-velocity diagram with normal pilot effort and skill. A minimum of five good runs should be flown by the FAA/AUTHORITY pilot at each altitude and weight. Runs by the company and FAA/AUTHORITY pilot may be averaged. Effects of missing the V_{50} speed by some amount (± 5 knots, for example) or other small changes in profile should be evaluated to determine if gross performance changes result from small piloting errors. Engine failures should be conducted along the takeoff profile to assure safe landing capability. Past programs have shown the low speed point immediately after addition of power to be particularly critical. Night takeoffs should at least be qualitatively evaluated to assure the takeoff procedures are compatible for night operation.

(3) Test Results. Test results are utilized in constructing the flight manual takeoff distance charts required by § 29.1587. The takeoff surface utilized in conducting these takeoff distance and engine failure tests should be included in the flight manual. The “climbout speed” should also be defined and included in the flight manual. The airspeed utilized at the 50-foot point in the conduct of these tests must be clearly defined to allow compliance with § 29.1323. Test results may be extrapolated in accordance with guidance contained in paragraph AC 29.45.

(4) Test Techniques. For the FAA/AUTHORITY test data runs which will result in rotorcraft flight manual (RFM) performance, only the operational cockpit instrumentation as shown on the minimum equipment list and the piloting procedures from the RFM should be used. A useful technique is to “lead” the targeted V_{50} speed by a fixed amount, so that a smooth, consistent, and operationally realistic transition may be made between the acceleration and climbout phases; e.g., begin rotation at 35 knots to achieve 46 knots passing 50 feet. This and other pertinent information defining the takeoff flight path are required flight manual entries per § 29.1587(b).

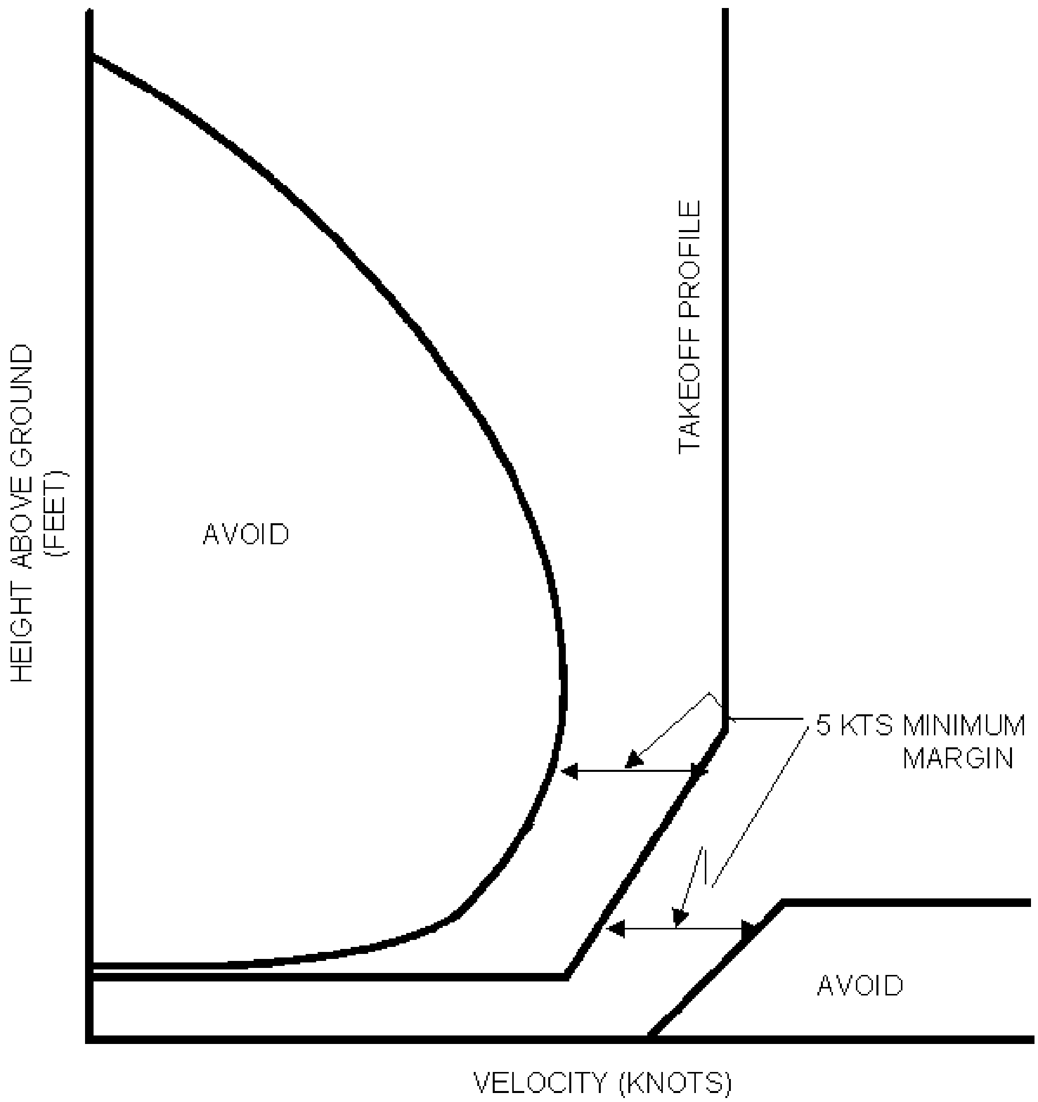


FIGURE AC 29.63-1 CONVENTIONAL TAKEOFF PROFILE CATEGORY B

AC 29.65. § 29.65 (Amendment 29-15) CLIMB: (ALL ENGINES OPERATING).

a. Explanation.

(1) Section 29.65 requires in part that the steady rate of climb be determined for each Category B rotorcraft with maximum continuous power on each engine for the range of weights, altitudes, and temperatures for which certification is requested. The climb airspeed should be the best rate-of-climb (V_Y) for standard day sea level conditions at maximum weight and at a speed(s) selected by the applicant for other conditions not to exceed V_{NE}. The applicant can either publish a climb schedule in accordance with the above or utilize a constant climb airspeed for all conditions. Equivalent levels of safety have been found wherein the applicant was allowed to select a climb airspeed that was not the actual V_Y. The selected airspeed must be consistent with the speed used to show compliance with such items as cooling, stability, etc. The rate-of-climb resulting from the selected climb airspeed versus that from the actual V_Y shall not differ to an extent that a pilot will be encouraged, by appreciable increases in climb performance to fly a climb airspeed different from that published in the Flight Manual.

(2) For Category A rotorcraft, if V_{NE} at any altitude is less than the maximum gross weight sea level standard day condition V_Y, the steady rate-of-climb must be determined at the climb speed(s) selected by the applicant not to exceed V_{NE}. The climb performance must be determined from 2,000 feet below the altitude from where V_{NE} intersects V_Y up to the maximum altitude for which certification is requested. This should be done utilizing maximum continuous power on each engine with the landing gear retracted.

b. Procedure to Determine V_Y.

(1) Sawtooth climbs may be used to determine the best rate-of-climb airspeed V_Y. If such a technique is used, climbs should be flown in pairs on opposite headings 90° to the winds at the test altitude. This procedure will minimize any windshear effects. All testing should be done in smooth air. Windshear is usually an indication of unstable air or a temperature inversion and should be avoided. The climbs are flown on reciprocal headings for approximately 5 minutes through a 1,000-foot band, or a comparable time/altitude band, using maximum continuous power at a constant airspeed. Periodic power adjustments may be necessary. Additional reciprocal heading climbs must also be conducted at different airspeeds sufficient to bracket the lowest point of the power required versus airspeed curve. This technique can be repeated at different altitudes to obtain V_Y throughout the altitude range.

(2) Level flight performance (speed power) may also be used to determine the best rate-of-climb airspeed (V_Y). The testing should be done in smooth air. The advantage of this method is that less time is required, and the accuracy is equivalent to the sawtooth climb method. The test can be repeated at various altitudes to determine the V_Y throughout the altitude range desired for the rotorcraft. The test at each altitude

should be conducted at a constant weight over sigma (W/σ). The test is normally started at the desired W/σ with maximum continuous power, or at V_{NE}, in level flight. A series of points should be taken, reducing airspeed 10 to 15 knots between points, with the lowest speed point at approximately 20 to 30 knots. Weight should be computed for each point and the test altitude adjusted to maintain a constant W/σ. After the data are reduced to standard day conditions, the minimum power required airspeed will be the V_Y speed.

(3) Prior to the flight test, the rotorcraft should be ballasted to the desired gross weight and the critical center of gravity. The airspeed should be stabilized prior to data acquisition. Data to be recorded includes time, altitude, airspeed, ambient temperature, engine parameters, torque(s), rotor RPM, fuel reading, aircraft heading, external configuration, etc. Power setting, weight, and climb airspeed should be planned prior to flight. For some turboshaft engines, temperature and/or engine speed limits may be reached prior to a limiting torque. The test team should verify that the resulting power utilized in these tests closely approximates the power producing capabilities of installed minimum specification engine.

c. Procedure to Determine all Engine Operating Climb Performance.

(1) Background. Continuous climbs are conducted at the appropriate climb airspeeds as outlined above in order to obtain the rotorcraft's climb performance for the flight manual. By-products are a qualitative evaluation of the rotorcraft handling characteristics in a climb and engine data to assist in the determination of installed power available.

(2) Techniques. The techniques used to determine this performance may be the same as those used in the V_Y determination. The climbs are conducted on reciprocal headings at the established airspeed(s) through the target altitude range. The same parameters are recorded. The rotorcraft will usually climb very rapidly during the first few thousand feet; therefore, the data acquisition method must be timely if accurate results are expected. This procedure is usually repeated at weight extremes. The resulting data must then be corrected for power and weight. Power and weight corrections are satisfactory, provided the test powers and weights closely approximate the target values to make the weight and power corrections accurate. Once this data is finalized and corrected for all the flight test variables, interpolation for intermediate weights can be made with a high degree of reliability. If the rotorcraft has any stability augmentation system, vent systems, etc., which may influence the climb performance, then it must be accounted for. Caution should be taken that anti-ice, air-conditioning, etc., are not on unless the performance is being established specifically for those conditions.

AC 29.65A (AC 29.64) §§ 29.64 and 29.65 (Amendment 29-39) CLIMB (GENERAL AND ALL ENGINES OPERATING).

a. Explanation.

(1) Amendment 29-39 relocated and clarified the general climb requirements into a new § 29.64 and added requirements to determine Category A climb performance in § 29.65. The guidance material presented in paragraph AC 29.67 does not apply to rotorcraft certified with Amendment 29-39 or later. Sections 29.64 and 29.65 require that the steady rate of climb be determined with maximum continuous power on each engine for the range of weights, altitudes, and temperatures for which certification is requested. The climb airspeed should be the best rate-of-climb (V_Y) for standard day sea level conditions at maximum weight and at a speed(s) selected by the applicant for other conditions not to exceed V_{NE}. The applicant can either publish a climb schedule in accordance with the above or utilize a constant climb airspeed for all conditions. Equivalent levels of safety have been found wherein the applicant was allowed to select a climb airspeed that was not the actual V_Y. The selected airspeed should be consistent with the speed used to show compliance with such items as cooling, stability, etc. The rate-of-climb resulting from the selected climb airspeed versus that from the actual V_Y shall not differ to an extent that a pilot will be encouraged by appreciable increases in climb performance to fly a climb airspeed different from that published in the Flight Manual.

(2) If V_{NE} at any altitude is less than the maximum gross weight sea level standard day condition V_Y, the steady rate-of-climb should be determined at the climb speed(s) selected by the applicant not to exceed V_{NE}. The climb performance should be determined from 2,000 feet below the altitude from where V_{NE} intersects V_Y up to the maximum altitude for which certification is requested. This should be done utilizing maximum continuous power on each engine with the landing gear retracted.

b. Procedure to Determine V_Y.

(1) Sawtooth climbs may be used to determine the best rate-of-climb airspeed V_Y. If such a technique is used, climbs should be flown in pairs on opposite headings 90° to the winds at the test altitude. This procedure will minimize any windshear effects. All testing should be done in smooth air. Windshear is usually an indication of unstable air or a temperature inversion and should be avoided. The climbs are flown on reciprocal headings for approximately 5 minutes through a 1,000-foot band, or a comparable time/altitude band, using maximum continuous power at a constant airspeed. Periodic power adjustments may be necessary. Additional reciprocal heading climbs should also be conducted at different airspeeds sufficient to bracket the lowest point of the power required versus airspeed curve. This technique can be repeated at different altitudes to obtain V_Y throughout the altitude range.

(2) Level flight performance (speed power) may also be used to determine the best rate-of-climb airspeed (V_Y). The testing should be done in smooth air. The advantage of this method is that less time is required, and the accuracy is equivalent to the sawtooth climb method. The test can be repeated at various altitudes to determine the V_Y throughout the altitude range desired for the rotorcraft. The test at each altitude should be conducted at a constant weight over sigma (W/σ). The test is normally

started at the desired W/σ with maximum continuos power, or at V_{NE}, in level flight. A series of points should be taken, reducing airspeed 10 to 15 knots between points, with the lowest speed point at approximately 20 to 30 knots. Weight should be computed for each point and the test altitude adjusted to maintain a constant W/σ. After the data are reduced to standard day conditions, the minimum power required airspeed will result in the airspeed for maximum rate of climb. However, aircraft stability may suggest that a higher climb speed may be used for $V_{Y.}$

(3) Prior to the flight test, the rotorcraft should be ballasted to the desired gross weight and the critical center of gravity. The airspeed should be stabilized prior to data acquisition. Data to be recorded includes time, altitude, airspeed, ambient temperature, engine parameters, torque(s), rotor RPM, fuel reading, aircraft heading, external configuration, etc. Power setting, weight, and climb airspeed should be planned prior to flight. For some turboshaft engines, temperature and/or engine speed limits may be reached prior to a limiting torque. The test team should verify that the resulting power utilized in these tests closely approximates the power producing capabilities of installed minimum specification engine.

c. Procedure to Determine all Engine Operating Climb Performance.

(1) Background. Continuous climbs are conducted at the appropriate climb airspeeds as outlined above in order to obtain the rotorcraft's climb performance for the flight manual. By-products are a qualitative evaluation of the rotorcraft handling characteristics in a climb and engine data to assist in the determination of installed power available.

(2) Techniques. The techniques used to determine this performance may be the same as those used in the V_Y determination. The climbs are conducted on reciprocal headings at the established airspeed(s) through the target altitude range. The same parameters are recorded. The rotorcraft will usually climb very rapidly during the first few thousand feet; therefore, the data acquisition method should be timely if accurate results are expected. This procedure is usually repeated at weight extremes. The resulting data should then be corrected for power and weight. Power and weight corrections are satisfactory, provided the test powers and weights closely approximate the target values to make the weight and power corrections accurate. Once this data is finalized and corrected for all the flight test variables, interpolation for intermediate weights can be made with a high degree of reliability. If the rotorcraft has any stability augmentation system, vent systems, etc., which may influence the climb performance, then it should be accounted for. Caution should be taken that anti-ice, air-conditioning, etc., are not on unless the performance is being established specifically for those conditions.

AC 29.67. § 29.67 (Amendment 29-34) CLIMB: ONE ENGINE INOPERATIVE.

a. Explanation.

(1) Section 29.67 requires that Category A rotorcraft must be capable of a steady rate-of-climb without ground effect, of at least 100 feet per minute for all combinations of weight, altitude, temperature, and center of gravity for which takeoffs are to be scheduled. The rate-of-climb is determined with the critical engine inoperative and the remaining engine(s) operating within approved operating limits. The landing gear is extended and the airspeed is the takeoff safety speed (V_{TOSS}) selected by the applicant.

(2) In addition, the steady rate-of-climb must be at least 150 feet per minute at 1,000 feet above the takeoff surface for which takeoffs are to be scheduled. The rate-of-climb will be determined with the critical engine inoperative and the remaining engine(s) at maximum continuous or the 30-minute minimum specification installed power available values. The landing gear is retracted and the airspeed is that selected by the applicant.

b. Procedures.

(1) One of the acceptable procedures used to obtain the required climb performance is similar to the all engine climb performance determination (paragraph AC 29.65) except that the V_{TOSS} and the Category A climb speed may be selected by the applicant for different weights and ambient conditions. The Category A climb speed could be a single speed, vary as V_Y does, or actually be V_Y. Making a Category A climbout speed equal to V_Y should be encouraged to simplify cockpit procedures. The required results are the allowable weight, altitude, and temperature combinations wherein the rotorcraft is capable of demonstrating 100 feet per minute rate-of-climb at V_{TOSS} and 150 feet per minute rate-of-climb at 1,000 feet above the takeoff surface. Either of these two climb requirements may establish the maximum allowable takeoff weight.

(2) For multiengine Category B rotorcraft with engine isolation, the steady rate of climb or descent must be determined at V_Y, using maximum continuous power and 30-minute power if that rating is approved. Appropriate performance data must be included in the Rotorcraft Flight Manual to cover variations in gross weight, altitude, and temperature.

(3) Since climb performance testing is normally conducted separately from Category A and B takeoff performance testing, it is imperative the engine power(s), rotor RPM, and aircraft configuration be the same as those used during the takeoff testing to ensure the climb performance demonstrated will be that attainable immediately after an engine failure during takeoff. The allowable pilot/crew actions during the Category A takeoff and climbout maneuver must be thoroughly evaluated. The pilot's full attention is required to control the rotorcraft during this phase of flight. Permitting the pilot to readjust (beep) the rotor RPM during this phase of flight should be considered only if such adjustment can be accomplished without a significant increase in pilot workload.

(4) A typical sequence for selecting the various speeds to comply with this requirement is as follows:

(i) Conduct sawtooth climbs at the various airspeeds (V_Y and below) up to the proposed takeoff and landing altitudes. From this a determination can be made regarding the maximum allowable weight that will result in a rate of climb of 150 feet per minute at the selected V_Y for the proposed ambient conditions.

(ii) At the same time determine the minimum value of V_{TOSS} that will result in 100 feet per minute rate of climb at the maximum weight determined in (b)(4)(i).

AC 29.67A. § 29.67 (Amendment 29-39) CLIMB: ONE ENGINE INOPERATIVE.

a. Explanation.

(1) Amendment 29-39 expanded the OEI rate of climb requirements. The guidance material presented in paragraph AC 29.67 does not apply to rotorcraft certified with Amendment 29-39 or later. Section 29.67 requires that Category A rotorcraft should be capable of a steady rate-of-climb without ground effect 200 feet above the takeoff surface, of at least 100 feet per minute for all combinations of weight, altitude, temperature, and center of gravity for which takeoffs are to be scheduled. The rate-of-climb is determined with the critical engine inoperative and the remaining engine(s) operating within approved operating limits. The landing gear is extended and the airspeed is the takeoff safety speed (V_{TOSS}) selected by the applicant.

(2) The steady rate-of-climb should be at least 150 feet per minute at 1,000 feet above the takeoff surface for which takeoffs are to be scheduled. The rate-of-climb will be determined with the critical engine inoperative and the remaining engine(s) at maximum continuous or the 30-minute minimum specification installed power available values. The landing gear is retracted and the airspeed is that selected by the applicant.

(3) Additionally, the steady state rate of climb or descent must be determined with the critical engine inoperative and the remaining engines at OEI maximum continuous power and at 30-minute OEI power if applicable. This performance must be scheduled throughout the ranges of weight, altitude and temperatures for which certification is requested with the landing gear retracted, at an airspeed selected by the applicant.

b. Procedures.

(1) One of the acceptable procedures used to obtain the required climb performance is similar to the all engine climb performance determination (paragraph AC 29.65) except that the V_{TOSS} and the Category A climb speed may be selected by the applicant for different weights and ambient conditions. The Category A climb speed could be a single speed, vary as V_Y does, or actually be V_Y. Making a Category A climbout speed equal to V_Y should be encouraged to simplify cockpit

procedures. The required results are the allowable weight, altitude, and temperature combinations wherein the rotorcraft is capable of demonstrating 100 feet per minute rate-of-climb at V_{TOSS} at a height of 200 feet above the takeoff surface and 150 feet per minute rate-of-climb at 1,000 feet above the takeoff surface. Either of these two climb requirements may establish the maximum allowable takeoff weight.

(2) For multiengine Category B rotorcraft with engine isolation, the steady rate of climb or descent should be determined at V_Y, using maximum continuous power, maximum continuous OEI power, and 30-minute power if that rating is approved. Appropriate performance data should be included in the Rotorcraft Flight Manual to cover variations in gross weight, altitude, and temperature.

(3) Since climb performance testing is normally conducted separately from Category A and B takeoff performance testing, it is imperative the engine power(s), rotor RPM, and aircraft configuration be the same as those used during the takeoff testing to ensure the climb performance demonstrated will be that attainable immediately after an engine failure during takeoff. The allowable pilot/crew actions during the Category A takeoff and climbout maneuver should be thoroughly evaluated. The pilot's full attention is required to control the rotorcraft during this phase of flight. Permitting the pilot to readjust (beep) the rotor RPM during this phase of flight should be considered only if such adjustment can be accomplished without a significant increase in pilot workload.

(4) A typical sequence for selecting the various speeds to comply with this requirement is as follows:

(i) Conduct sawtooth climbs at the various airspeeds (V_Y and below) up to the proposed takeoff and landing altitudes. From this, a determination can be made regarding the maximum allowable weight that will result in a rate of climb of 150 feet per minute at the selected V_Y for the proposed ambient conditions.

(ii) At the same time, determine the minimum value of V_{TOSS} that will result in 100 feet per minute rate of climb at the maximum weight determined in b(4)i.

AC 29.71. § 29.71 (Amendment 29-12) ROTORCRAFT ANGLE OF GLIDE: CATEGORY B.

a. Explanation.

(1) Performance capabilities during stabilized autorotative descent are useful pilot tools to assist in the management of a Category B rotorcraft when all engines fail. This information is also useful in determining the suitability of available landing areas along a given route segment.

(2) Two speeds are of particular importance, the speed for minimum rate of descent and the speed for best angle of glide. These speeds are required as flight manual entries per § 29.1587. The speed for minimum rate of descent is useful for

engine failure conditions at higher altitudes and the pilot is required to perform some time-related task, engine restart, float inflation, radio calls, etc. The speed for best angle of glide is a somewhat higher speed that is of particular use when it is necessary to reach a distant landing area. This speed, with appropriate rotor RPM, provides the maximum horizontal distance available from a particular altitude assuming zero wind conditions.

(3) A third speed, recommended autorotation speed, may be provided in addition to minimum rate of descent speed and maximum glide angle speed. The recommended speed for autorotation is usually optimized to assure an effective flare capability and yet be slow enough to allow a controlled, relatively slow touchdown condition. Recommended autorotation speed is ordinarily between the minimum rate of descent and maximum glide angle speeds. The recommended autorotation speed may be provided in the Rotorcraft Flight Manual. The relationship between minimum rate of descent, best glide angle, and recommended autorotation speed is shown in figure AC 29.71-1.

(4) Forward center of gravity is usually critical, however, center of gravity effects should be spot-checked to confirm this for a given design.

b. Procedures.

(1) Tests are conducted at speeds which bracket the anticipated speeds for minimum rate of descent and best glide angle. On a power required plot, the speed for minimum power required approximates the speed for minimum rate of descent. The speed for maximum range glide may be estimated by drawing a tangent from the origin to the power required curve.

(2) Autorotative performance tests may be conducted in conjunction with the climb performance tests. The required data are similar for both tests and it is sometimes convenient and efficient to run alternating climbs and descents through a desired altitude band. Descents should be conducted on reciprocal headings and results averaged in the same manner as climb performance tests.

(3) A reduction in rotor RPM from the normal power-on value may enhance autorotative performance. If the applicant wishes to develop autorotative performance at RPM values significantly below the governing or power-on range, the practicality of reducing and controlling RPM at the lower value and of then increasing RPM as a landing is approached, must be considered. At low weights and low density altitudes, full down collective may automatically produce lower RPM values and this condition is, of course, acceptable provided the approved power-off RPM range is not exceeded.

(4) Care must be taken to make certain that no engine power is delivered to the rotor drive system since a very small amount of power can have a large effect on descent performance.

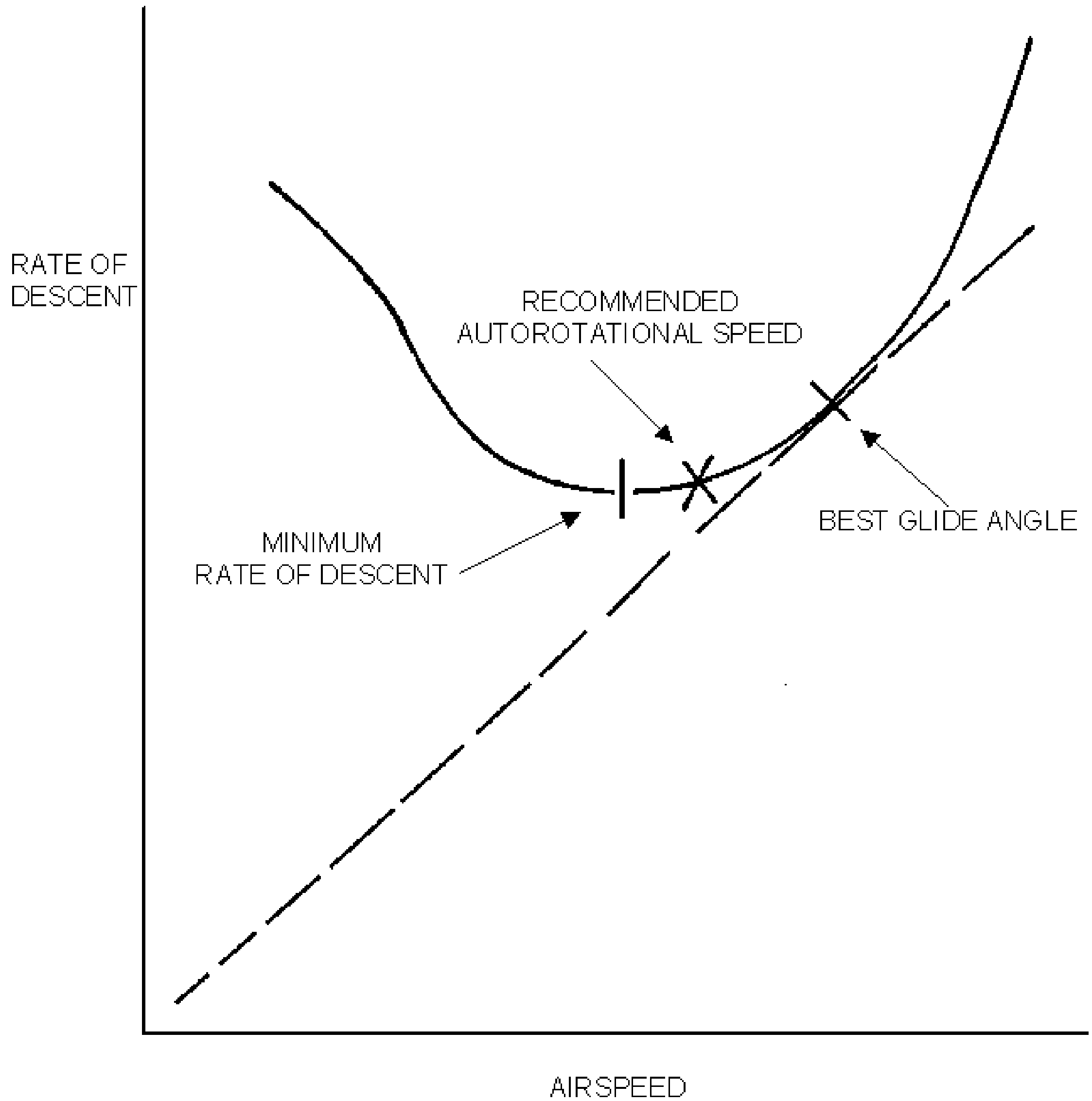


FIGURE AC 29.71-1 AUTOROTATIONAL CHARACTERISTICS - TYPICAL

AC 29.73. § 29.73 (Amendment 29-3) PERFORMANCE AT MINIMUM OPERATING SPEED. HOVER PERFORMANCE FOR ROTORCRAFT.

(For performance at minimum operating speed and for hover performance after Amendment 38, see § 29.49 and paragraph AC 29.49).

a. Explanation.

(1) For the purpose of this manual, the word "hover" applies to a rotorcraft that is airborne at a given altitude over a fixed geographical point regardless of wind. Pure hover is accomplished only in still air.

(2) The regulatory requirement for hover performance, § 29.73, refers to hover in ground effect (IGE). For some applications, such as external load operations, hover performance out-of-ground effect (OGE) is necessary; however, it is not required by this section. Hover OGE is that condition, where an increase in height above the ground will not require additional power to hover. Hover OGE is the absence of measurable ground effect. It can be less than one rotor diameter at low gross weight increasing significantly at high gross weights. The lowest OGE hover height at gross weight may be approximated by placing the lowest part of the vehicle 1 ½ rotor diameters above the surface.

(3) The objective of hover performance tests is to determine the power required to hover at different gross weights, ambient temperatures, and pressure altitudes. Using nondimensional power coefficients (C_P) and thrust coefficients (C_T) for normalizing and presenting test results, a minimum amount of data are required to cover the rotorcraft's operating envelope.

(4) Hover performance tests must be conducted over a sufficient range of pressure altitudes and weights to cover the approved ranges of those variables for takeoff and landings. Additional data should be acquired during cold ambient temperatures, especially at high altitudes, to account for possible Mach effects.

(5) The minimum hover height for which data should be obtained and subsequently presented in the flight manual should be the same height consistent with the minimum hover height demonstrated during the takeoff tests. Refer to paragraph AC 29.51 for the procedure to determine the minimum allowable hover height.

b. Procedures.

(1) Two methods of acquiring hover performance data are the tethered and free flight techniques. The tethered technique is accomplished by tethering the rotorcraft to the ground using a cable and load cell. The load cell and cable are attached to the ground tie-down and to the rotorcraft cargo hook. The load cell is used to measure the rotorcraft's pull on the cable. Hover heights are based on skid or wheel height above

the ground. During tethered hover tests, the rotorcraft should be at light gross weight. The rotorcraft will be stabilized at a fixed power setting and rotor speed at the appropriate skid or wheel height. Once the required data are obtained, power should be varied from the minimum to the maximum allowed at various rotor RPM. This technique will produce a large C_T/C_P spread. The load cell reading is recorded for each stabilized point. The total thrust the rotor produces is the rotorcraft's gross weight, weight of the cables and load cell plus cable tension. Care must be taken that the cable tension does not exceed the cargo hook limit or load capacity of the tie-down. For some rotorcraft, it may be necessary to ballast the rotorcraft to a heavy weight in order to record high power hover data.

(2) The pilot maintains the rotorcraft in position so that the cable and load cell are perpendicular to the ground. To insure the cable is vertical, two outside observers, one forward of the rotorcraft and one to one side, can be used. Either hand signals or radio can be used to direct the pilot. The observers should be provided with protective equipment. This can also be accomplished by attaching two accelerometers to the load cell which sense movement along the longitudinal and lateral axes. Any displacement of the load cell will be reflected on instrumentation in the cockpit and by reference to this instrumentation, the rotorcraft can be maintained in the correct position. Increased caution should be utilized as tethered hover heights are decreased because the rotorcraft may become more difficult to control precisely. The tethered hover technique is especially useful for OGE hover performance data because the rotorcraft's internal weight is low and the cable and load cell can be jettisoned in the event of an engine failure or other emergency.

(3) To obtain consistent data, the wind velocity should be less than 3 knots or less as there are no accurate methods of correcting hover data for wind effects. Large rotorcraft with high downwash velocities may tolerate higher wind velocities. The parameters usually recorded at each stabilized condition are:

(i) Engine torques.

(ii) Rotor speed.

(iii) Ambient temperatures.

(iv) Pressure altitude.

(v) Fuel used (or remaining).

(vi) Load cell reading.

(vii) Generator(s) load.

As a technique, it is recommended the rotorcraft be loaded to a center of gravity near the hook to minimize fuselage angle changes with varying powers. All tethered hover data should be verified by a limited spotcheck using the free flight technique. The free flight technique as contained in paragraph b(4) below will determine if any problems, such as load cell malfunctions have occurred. The free flight hover data must fall within the allowable scatter of the tethered data.

(4) If there are no provisions or equipment to conduct tethered hover tests, the free flight technique is also a valid method. The disadvantage of this technique as the primary source of data acquisition is that it is very time consuming. In addition a certain element of safety is lost OGE in the event of emergency. The rotorcraft must be reballasted to different weights to allow the maximum C_T/C_P spread. When using the free flight technique, either as a primary data source or to substantiate the tethered technique, the same considerations for wind, recorded parameters, etc., as used in the tethered technique apply. Free flight hover tests should be conducted at CG extremes to verify any CG effects. If the rotorcraft has any stability augmentation system which may influence hover performance, it must be accounted for.

(5) It is extremely difficult to determine when a rotorcraft is hovering OGE at high altitudes above ground level since there is no ground reference. In a true hover, the rotorcraft will drift with the wind. Numerous techniques have been tried to allow OGE hover data acquisition at high altitudes, all of which have resulted in much data scatter. Until a method is proposed and found acceptable to the FAA/AUTHORITY, OGE hover data must be obtained at the various altitude sites where IGE hover data is obtained. Hover performance can usually be extrapolated up to a maximum of 4,000 feet.

AC 29.75. § 29.75 (Amendment 29-17) LANDING.

a. Explanation.

(1) This rule incorporates all of the landing performance requirements for transport category rotorcraft. It consolidates requirements for landing data, Category A landing, Category A flight data, and Category B landing. Parallel takeoff requirements are located in four separate sections of the rule, §§ 29.51 through 29.63. As such, to assure necessary subjects are treated separately, the following discussion will be separated into three parts: (a) a general discussion of basic landing distance requirements, (b) Category A requirements (including vertical landing), and (c) Category B requirements.

(2) All landing performance data are corrected to a smooth, dry, hard, level landing surface condition. As with other flight maneuvers, landings must be accomplished with acceptable flight and ground characteristics using normal pilot skills. The rule states that Category A and B landing data must be determined at each approved WAT (Weight, Altitude, Temperature) condition. Reasonable sampling and extrapolation methods are, of course, allowed. General guidance on those subjects is given in paragraph AC 29.45. As in other performance areas, engines must be

operated within approved limits. Power considerations are the same as those described under paragraph b(2)(ii)(C).

(3) Unlike fixed wing aircraft, rotorcraft typically require significantly more landing surface area with an engine inoperative than with all engines operating. Because of this characteristic, the landing distance requirements are met with at least one engine inoperative to assure the most conservative landing distance measurement is achieved.

b. Procedures.

(1) Category A Requirements.

(i) Explanation. The Category A certification concept limits landing weight to a value that will allow the rotorcraft, following an engine failure at the landing decision point (LDP), to land within the available runway or to execute a balked landing, descending no lower than 35 feet above the landing surface. See figure AC 29.75-1.

(A) LDP. The Category A landing profile begins with an assumed engine failure at or prior to the LDP. The LDP is typically defined in terms of airspeed, rate of descent, and altitude above the landing surface. The approach path angle can be defined by LDP airspeed and rate of descent values. Definition of the LDP should include an approach angle because both the landing distance and the missed approach path are significantly influenced by landing approach angle. At any point in the single engine approach path down to and including the LDP, the pilot may elect to land or to execute a balked landing and he is assured both an adequate surface area for OEI landing and adequate climb capability for an OEI balked landing. Said another way, if an engine fails at any point down to and including the LDP, the pilot may safely elect to land or to “go around” by executing a balked landing. The LDP must be defined to permit acceleration to V_{TOSS} at an altitude no lower than 35 feet above the landing surface. The LDP represents a “commit” point for landing. Prior to the LDP in the one engine inoperative approach, the pilot has a choice, he may either land or fly away. After passing the LDP he no longer has sufficient energy to assure transition to a balked landing condition without contacting the landing surface. If an engine fails after LDP in a normal (all engine) landing the pilot is committed to land. The LDP and landing approach path must be defined such that the critical areas of the height-velocity diagram are avoided. A typical LDP for conventional Category A profiles is 100 feet above the landing surface. LDP should be specified in terms of both actual altitude above the landing surface and indicated barometric altitude. Speed at the LDP should be specified in terms of indicated airspeed.

(B) Landing distance. Approach and landing path requirements are stated in general terms in paragraphs (b)(2) and (4) of § 29.75. The approach path must allow smooth transition for one engine inoperative landing and for balked landing maneuvers and must allow adequate clearance from potentially hazardous HV combinations. Paragraph (b)(4)(ii) implies that a less restrictive HV envelope may exist for the Category A approach condition in comparison to that determined under high power conditions in § 29.79. The manufacturer may elect to use this added capability. The added capability arises from the fact that lower power levels, a lower collective setting, and an established rate of descent accompany typical approach conditions as opposed to the more critical high power conditions of § 29.79. Landing distance is measured from a point 50 feet (25 feet for VTOL) above the landing surface to a stop. For flight manual purposes, the distance is from the point at which the lowest part of the rotorcraft first reaches 50 feet (25 for VTOL) to the foremost point of the rotorcraft (including rotor tip path) after coming to a stop.

(C) All engine out landing. Section 29.75(b)(5) contains the Category A certification requirement for “last” engine failure and all engine inoperative landing. The rule states that it must be possible to make a safe landing on a prepared surface after complete power failure during normal cruise. It is not intended that all engines be failed simultaneously. See paragraph AC 29.143a(2)(iii)(A) for the Category A sequential engine failure criteria. The conditions for last engine failure are maximum continuous power or 30-minute power if that rating is approved, “wings” level flight, and sudden engine failure with a pilot delay of 1 second or normal pilot recognition time, whichever is greater. Complete power failure has occurred in twin engine Category A rotorcraft. This requirement ensures that in the event of cockpit mismanagement, fuel exhaustion, improper maintenance, fuel contamination, or unforeseen mechanical failures, a safe autorotation entry can be made and a safe power-off landing can be affected. Two separate aspects of this rule are normally evaluated at different times during the test program. The last engine failure is normally evaluated during cruise or V_{NE} engine failure testing where instrumentation and critical loading have been established for those test conditions. See discussion under paragraph AC 29.143. The all engine out landing is ordinarily conducted in conjunction with an HV or Category A landing distance phase where ground instrumentation and safety equipment are available. The rotorcraft must be capable of conducting the all engine out landing at the takeoff and landing WAT limiting conditions up to the maximum altitude approved for takeoff and landing.

(ii) Procedures.

(A) Instrumentation/Equipment. Instrumentation requirements are basically the same as those for Category A takeoff. A photo theodolite, grid camera, or other position measuring equipment is needed, along with a ground station to measure wind, OAT, and humidity (if applicable). A two-way communication system between the aircraft and the position measuring equipment is essential. Aircraft instrumentation should include engine and flight parameters, control positions, power lever position, landing gear loads, and a method for synchronizing power cuts between the external light normally used for photo theodolite or camera, and onboard instrumentation. A record of rotor RPM at touchdown is necessary to assure it does not exceed transient limits. Rotor RPM at touchdown may be lower than the minimum transient limit for flight, provided stress limits are not exceeded. A crash recovery team with support of a fire engine is highly desirable.

(B) Establishing the LDP.

(1) Unless the rotorcraft is capable of hovering with one engine inoperative at the desired Category A weight, the LDP becomes largely a function of the runway length required for landing. If landing conditions to be scheduled include considerable runway length (on the order of 1,000 feet) the LDP may be defined at a relatively high speed allowing transition to a takeoff safety speed near V_Y which will allow the maximum amount of weight for compliance with the balked landing climb requirements of § 29.77(b)/§ 29.67(a)(1). In this case, the requirements of § 29.67(a)(2) usually become limiting. If the runway length is small, LDP will typically be at a lower speed and may be at a higher altitude to allow balked landing transition within the available distance. Landing weight may need to be reduced to allow landing from the lower speed or higher altitude decision point for shorter landing distances. In this case the requirements of § 29.67(a)(1) may be limiting. The climb performance and climb speeds required by § 29.67(a)(1) and (2) should be established prior to Category A landing tests.

(2) The one engine inoperative landing is similar in many respects to the height-velocity tests described in paragraph AC 29.79. Most of the comments, cautions, and techniques for HV also apply here even though typical flight conditions at LDP are less critical than limiting HV points due to a lower power level and an established rate of descent. The approach is made at a predetermined speed and one engine is made inoperative prior to LDP After the LDP, speed is reduced and the rotorcraft is flared to a conventional one engine inoperative landing. Depending on the landing characteristics and landing profile, the flare may be initiated either prior or subsequent to the 50-foot elevation utilized in determining landing distance. Testing should include an engine failure at the LDP with a 1-second pilot delay to assure safe landing capability for this critical case. A minimum of five acceptable runs for distance should be flown by the FAA/AUTHORITY pilot. These may be averaged with an equal number of acceptable runs by the company pilot.

(3) The balked landing portion of the landing profile is addressed under § 29.77, Balked Landing: Category A. For an explanation of that requirement and a discussion of those test procedures refer to paragraph AC 29.71.

(C) Power. Power should be limited to minimum specification values on the operating engine(s). This may be accomplished by adjustment of engine topping to minimum specification values for the range of atmospheric variables to be approved. This is frequently done by installing an adjustable device in the throttle linkage with a control in the cockpit so that engine topping can be accurately adjusted for varying ambient conditions. With such a device in the control system it becomes vitally important to check topping power prior to each test sequence.

(D) Aircraft Loading. Aft center of gravity is usually most critical for landing distance determination because visibility constraints limit the degree to which the pilot can flare the rotorcraft for landing. If a weight effect is shown, a minimum of two weights should be flown at each test altitude. One weight should be the maximum weight for prevailing conditions and the other should provide a sufficient spread to validate weight accountability.

(E) Extrapolation. Weight cannot be extrapolated above test weight. See discussion under Height-Velocity Testing in paragraph AC 29.79. If no marginal areas are apparent and an acceptable analytical method is used, performance data may be extrapolated ±4,000 feet density altitude from test conditions. (See paragraph AC 29.45.)

(F) Ambient Conditions. Appropriate test limits for ambient conditions such as wind and temperature are contained in paragraph AC 29.45. Test data must be corrected for existing wind conditions during landing distance tests. Credit for headwind conditions may be given during flight manual data expansion. Paragraph AC 29.45 details allowable wind credit.

(G) All engine out landing.

(1) Several procedures can be utilized to demonstrate compliance with the all engine out landing requirement. As discussed in the explanation portion of this paragraph, § 29.75(b) contains two separate requirements. One is the ability to transition safely into autorotation after failure of the last operative engine. This requirement is discussed in paragraph AC 29.143. The second aspect of this rule requires that a landing from autorotation be possible on a prepared surface. The second requirement is discussed below. The maneuver is entered by smoothly reducing power at an optimum autorotation airspeed at a safe height above a prepared landing surface. If a complete company test program has documented an all engine out landing to the GW/σ (gross weight/density ratio) limit for takeoff and landing at each altitude, verification tests may be initiated at those limiting weight conditions. If not, buildup testing should be initiated at light weight. This test is ordinarily conducted at mid center of gravity. Typically, all altitudes may be approved with two weight limit landings: one at sea level and one near maximum takeoff and landing altitude.

(2) Demonstrated compliance with this requirement is intended to show that an autorotative descent rate can be arrested, and forward speed at touchdown can be controlled to assure a reasonable chance of survivability for the all engine failure condition. The touchdown speed (less than 40 KIAS is recommended) should be consistent with the type design limits including landing gear capability, aircraft visibility, and any other factors affecting repeatability of the maneuver. On Category A rotorcraft, rotor inertia is typically much lower than for single engine rotorcraft. RPM decays rapidly when the last engine is made inoperative. Also, due to this relatively low inertia level, considerable collective may be needed to prevent rotor overspeed conditions when the rotorcraft is flared for landing. Also, when testing final maximum weight points, the pilot should anticipate a need for considerable collective pitch to control rotor overspeed during autorotative descent, particularly at high altitude WAT limiting conditions. Some designs incorporate features which may lead to rotorcraft damage in

testing this requirement (e.g., droop stop breakage or loss of directional control with skids) if landings are conducted to a full stop with the engines cut off.

(3) The intent of this rule is to demonstrate controlled touchdown conditions and freedom from loss of control or apparent hazard to occupants when landing with all engines failed. In these cases compliance can be demonstrated by leaving throttles in the idle position and assuring no power is delivered to the drive train. Also, computer analysis may be used in conjunction with simulated in-flight checks to give reasonable assurance that an actual safe touchdown can be accomplished. Another method may be to make a power recovery after flare effectiveness of the rotorcraft has been determined. Other methods may be considered if they lead to reasonable assurance that descent can be arrested and forward speed controlled to allow safe landing with no injury to occupants when landing on a prepared surface with all engines failed. Regardless of the method(s) used to comply with this requirement, careful planning and analyses are very important due to the potentially hazardous aspects of power off simulation and landing of a Category A rotorcraft totally without power. Considerations for weight and altitude extrapolation are the same as those for HV testing (reference paragraph AC 29.79.) The all-engine-inoperative landing test is ordinarily done in conjunction with height velocity tests because ground and onboard instrumentation requirements are the same for both tests.

(H) Vertical Landings. The reader should be familiar with the preceding discussion of conventional Category A landing profiles because duplicate information is not repeated here. A typical vertical landing profile is shown in figure AC 29.75-2. This profile is equally applicable to both ground level and pinnacle sites. The profile begins at a stabilized single engine approach condition. It must be possible to make a safe OEI landing or go-around at any point prior to the LDP. At the LDP the aircraft becomes committed to landing. A safe landing must be possible in case of an engine failure at any point before or after the LDP. Testing should include a simulated failure at LDP with a 1-second delay or normal pilot response time, whichever is longer, and subsequent landing within the allowable area. The LDP is typically well above the 25-foot point from which landing distance is measured. The landing distance is the distance from the point at which the lowest portion of the rotorcraft reaches 25 feet above the landing surface to the forward-most point after coming to a stop (including main rotor tip path). The LDP becomes very important for landing on small, elevated heliports. The LDP must be clearly defined and flight manual instructions should carefully explain any pilot procedures. An illustration similar to figure AC 29.75-2 with somewhat more detailed information is most useful. Night OEI landings should be conducted to verify suitable visibility for both internal and external vertical landing cues.

c. Category B Requirements.

(1) Explanation. Section 29.75(c) contains the Category B landing requirements. For rotorcraft that do not meet the Category A powerplant installation requirements of this part, landing tests are conducted with all engines inoperative in an autorotative descent condition. Landing distance is measured from the 50-foot point to the point at which the rotorcraft is completely stopped (approximately 3 knots for water

landings). The autorotative approach speed is selected by the applicant. The landing maneuver is similar to that referred to during normal training flights as a practice autorotation. As in HV tests, care must be taken to assure no power is delivered to the rotor drive system during these tests. A small amount of power can have a significant effect on landing test results. Multiengine rotorcraft incorporating Category A engine isolation features may conduct landing distance tests with only one engine inoperative using the procedures prescribed above for Category A. For these rotorcraft the one engine inoperative condition typically results in much shorter distances due both to a much lower speed at the 50-foot point and the added power available for flaring and cushioning the landing. Instrumentation requirements are the same as those described under Category A above. Appropriate ambient conditions and allowable extrapolation are discussed under paragraph AC 29.45.

(2) Procedures. Prior to conducting these tests the crew should be familiar with the engine inoperative landing characteristics of the rotorcraft. For Category B rotorcraft without engine isolation, the flight profile may be entered in the same manner as a straight-in practice autorotation. It is recommended that for safety reasons idle power be used if a “needle split” (no engine power to the rotor) can be achieved. In some cases, a low engine idle adjustment has been set to assure needle split is attained. In other cases a temporary detent between idle and cutoff was used on the throttle. In a third case the engine was actually shut down on sample runs to verify that the engine power being delivered was not materially influencing landing capability or landing distances. The landing flare may be initiated prior to the 50-foot point. The flare is maintained as long as is reasonable to dissipate speed and build RPM. Rotor RPM must stay within allowable limits. Aft center of gravity is ordinarily critical due to visibility and flare-ability. Following the flare, the rotorcraft is allowed to touchdown in a landing attitude. Rotor RPM at touchdown should be recorded and it must be within allowable structural limits. For wheeled rotorcraft, the brakes are applied to an incipient skid for most efficient stopping. For rotorcraft on skids, the collective should be lowered as soon as characteristics allow in order to place a greater weight on the landing skids. These procedures would be appropriate flight manual entries to show how landing distances can be realized. For flight manual purposes the landing distance should include the horizontal distance from the point at which the lowest part of the rotorcraft first reaches 50 feet above the landing surface to the point at the foremost part of the rotorcraft (including rotor tip path) after coming to a stop. For Category B rotorcraft with engine isolation, the landing procedures are as described for Category A landing. When conducting Category B landings utilizing Category A “procedures,” § 29.75(b)(2) can be misleading. No transition capability to balked landing is intended for Category B rotorcraft. Section 29.77, Balked Landing, Category A, applies only to Category A rotorcraft and not to Category B rotorcraft which incorporates Category A “design” features. Five acceptable landing runs should be flown by the FAA/AUTHORITY pilot at each test weight. Results may be averaged with an equal number of company runs. If a weight effect on landing distance is to be shown, a minimum of two weight extremes are normally tested.

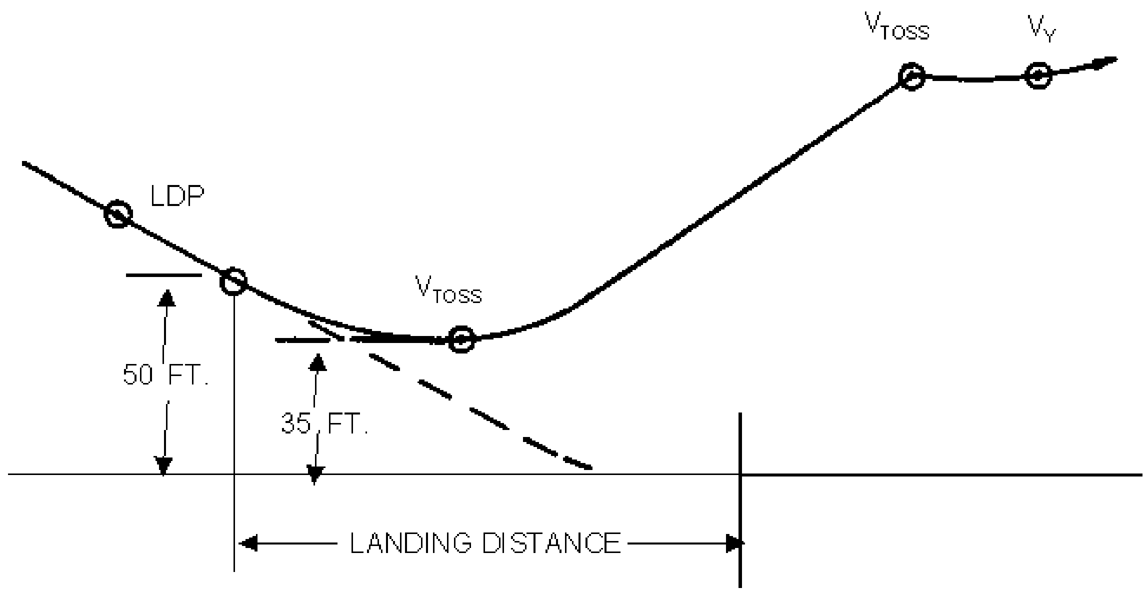


FIGURE AC 29.75-1 CATEGORY A CONVENTIONAL LANDING - CLEAR HELIPORT

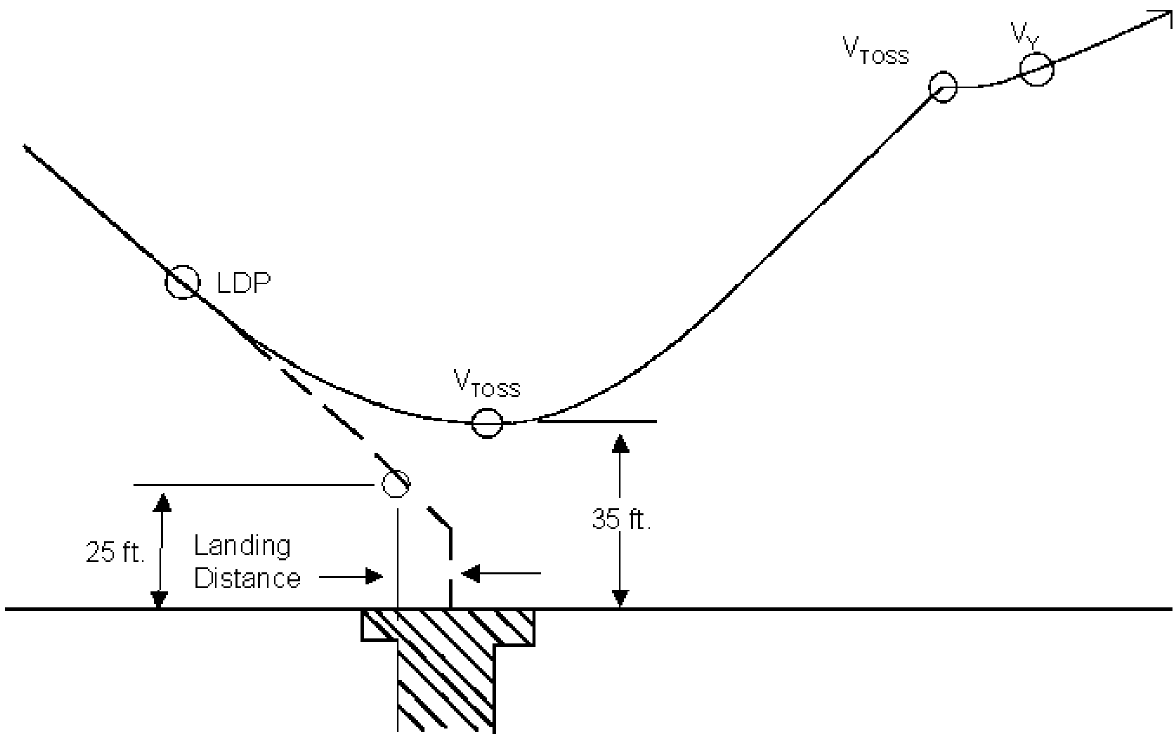


FIGURE AC 29.75-2 CATEGORY A VERTICAL LANDING

AC 29.75A. (AC's 29.77, 29.79, 29.81, & 29.83) §§ 29.75, 29.77, 29.79, 29.81, and 29.83 (Amendment 29-39) LANDING.

(For § 29.77 and § 29.79 prior to Amendment 39, see paragraphs AC 29.77 and AC 29.79 respectively.)

a. Explanation.

(1) Amendment 29-39 revised and relocated many of the landing requirements of Part 29. Changes were made to the general landing requirements of § 29.75. New requirements were added for designating a landing decision point (LDP) in § 29.77. The original § 29.79 was redesignated as a new § 29.87. Category A landing requirements were established in a new § 29.79. Requirements were added to determine landing distances in a new § 29.81. Revised Category B landing requirements were relocated from § 29.75(c) into a new § 29.83. The guidance material from paragraph AC 29.75 does not apply to rotorcraft certified with Amendment 29-39 or later.

(2) These rules incorporate all of the landing performance requirements for transport category rotorcraft. They contain the requirements for landing data, Category A landing, and Category B landing. Parallel takeoff requirements are located in eight separate sections of the rule, §§ 29.51 through 29.63. As such, to ensure that necessary subjects are treated separately, the following discussion will be separated into three parts: (a) a general discussion of basic landing distance requirements, (b) Category A requirements (including vertical landing), and (c) Category B requirements.

(3) All landing performance data are corrected to a smooth, dry, hard, level landing surface condition. As with other flight maneuvers, landings should be accomplished with acceptable flight and ground characteristics using normal pilot skills. The rule states that Category A and B landing data should be determined at each approved WAT (Weight, Altitude, Temperature) condition. Reasonable sampling and extrapolation methods are, of course, allowed. General guidance on those subjects is given in paragraph AC 29.45. As in other performance areas, engines should be operated within approved limits. Power considerations are the same as those described under paragraph b(1)(ii)(C).

(4) Unlike fixed-wing aircraft, rotorcraft typically require significantly more landing surface area with an engine inoperative than with all engines operating. Because of this characteristic, the Category A landing distance requirements are met with at least one engine inoperative to ensure the most conservative landing distance measurement is achieved.

b. Procedures - Category A Requirements.

(1) Explanation. The Category A certification concept limits landing weight to a value that will allow the rotorcraft, following an engine failure at the landing decision point (LDP), to land within the available area or to execute a balked landing descending no lower than 15 feet (or higher depending on rotorcraft geometry and performance characteristics) above the landing surface. For elevated heliports the rotorcraft may descend below the landing surface, but all parts of the rotorcraft must clear the heliport and other obstacles by not less than 15 feet. These minimum heights should be demonstrated with variations in piloting techniques and with pilot recognition and reaction times for engine failures occurring before/after LDP. See figure AC 29.75A-1.

(i) LDP. The Category A landing profile begins with an assumed engine failure at or prior to the LDP. The LDP is typically defined in terms of airspeed, rate of descent, and altitude above the landing surface. The approach path angle can be defined by LDP airspeed and rate of descent values. Definition of the LDP should include an approach angle because both the landing distance and the missed approach path are significantly influenced by landing approach angle. At any point in the single engine approach path down to and including the LDP, the pilot may elect to land or to execute a balked landing and he is assured both an adequate surface area for OEI landing and adequate climb capability for an OEI balked landing. Said another way, if an engine failure is recognized at any point down to and including the LDP, the pilot may safely elect to land or to “go-around” by executing a balked landing. The LDP should be defined to permit acceleration to V_{TOSS} clearing the landing surface by a minimum of 15 feet. The LDP represents a “commit” point for landing. Prior to the LDP in the one engine inoperative approach, the pilot has a choice, he may either land or fly away. After passing the LDP, he no longer has sufficient energy to assure transition to a balked landing condition without contacting the landing surface. If an engine failure is recognized after LDP in a normal (all engine) landing, the pilot is committed to land. The LDP and landing approach path should be defined such that critical areas of the height-velocity diagram are avoided. A typical LDP for conventional Category A profiles is 100 feet above the landing surface. LDP should be specified in terms of both actual height above the landing surface and indicated barometric altitude. Speed at the LDP should be specified in terms of indicated airspeed. The applicant may elect to develop an alternate all-engines-operating (AEO) approach procedure which meets the performance after engine failure requirements to execute a go-around before LDP or land after LDP but which could not be executed with OEI following an en route engine failure. If such alternate AEO procedures are provided, the Flight Manual should include the appropriate limitations prohibiting use of the AEO procedures after an en route engine failure. For such alternate AEO approach procedures it should be possible to execute a go-around and use the OEI approach procedure if the landing weight is consistent with such approach (the Flight Manual should indicate this OEI approach procedure and corresponding landing weight).

(ii) Landing distance. Approach and landing path requirements are stated in §§ 29.79(a)(2) and 29.83(a)(2). For Category A rotorcraft, the approach path should allow smooth transition for one-engine inoperative landing and for balked landing maneuvers. For all rotorcraft, the approach and landing paths should allow adequate clearance from potentially hazardous HV combinations. Landing distance is measured from a point 50 feet above the landing surface to a stop. For RFM presentation, the distance is from the aft most portion of the rotorcraft at the point at which the lowest part of the rotorcraft first reaches 50 feet to the foremost point of the rotorcraft (including rotor tip path) after coming to a stop.

(iii) All Engine Out Landing. § 29.79(b) contains the Category A certification requirement for an all engine inoperative landing. The rule states that it should be possible to make a safe landing on a prepared surface after complete power failure during normal cruise. It is not intended that all engines be failed simultaneously. See paragraph AC 29.143a(2)(iii)(A) for the Category A sequential engine failure criteria. The conditions for last engine failure are maximum continuous power or 30-minute power if that rating is approved, "wings" level flight, and sudden engine failure with a pilot delay of 1 second or normal pilot recognition time, whichever is greater. Complete power failure has occurred in twin engine Category A rotorcraft. This requirement ensures that in the event of cockpit mismanagement, fuel exhaustion, improper maintenance, fuel contamination, or unforeseen mechanical failures, a safe autorotation entry can be made and a safe power-off landing can be effected. Two separate aspects of this rule are normally evaluated at different times during the test program. The last engine failure is normally evaluated during cruise or VNE engine failure testing where instrumentation and critical loading have been established for those test conditions. See discussion under paragraph AC 29.143. The all engine out landing is ordinarily conducted in conjunction with an HV or Category A landing distance phase where ground instrumentation and safety equipment are available. The rotorcraft should be capable of conducting the all engine out landing at the takeoff and landing WAT limiting conditions up to the maximum altitude approved for takeoff and landing.

(2) Procedures.

(i) Instrumentation/Equipment. Instrumentation requirements are basically the same as those for Category A takeoff. A photo theodolite, grid camera, GPS, or other position measuring equipment is needed, along with a ground station to measure wind, OAT, and humidity (if applicable). A two-way communication system between the aircraft and the position measuring equipment is essential. Aircraft instrumentation should include engine and flight parameters, control positions, power lever position, landing gear loads, and a method for synchronizing aircraft position when the power is cut with onboard instrumentation. A record of rotor RPM at touchdown is necessary to ensure it does not exceed transient limits. Rotor RPM at touchdown may be lower than the minimum transient limit for flight, provided stress limits are not exceeded. A crash recovery team with support of a fire engine is highly desirable.

(ii) Establishing the LDP.

(A) Unless the rotorcraft is capable of hovering with one engine inoperative at the desired Category A weight, the LDP becomes largely a function of the runway length required for landing. If landing conditions to be scheduled include considerable runway length (on the order of 1,000 feet), the LDP may be defined at a relatively high speed allowing transition to a takeoff safety speed near V_Y which will allow the maximum amount of weight for compliance with the balked landing climb requirements of § 29.85(b)/§ 29.67(a)(1). In this case, the requirements of § 29.67(a)(2) usually become limiting. If the runway length is small, LDP will typically be at a lower speed and may be at a higher altitude to allow balked landing transition within the available distance. Landing weight may need to be reduced to allow landing from the lower speed or higher altitude decision point for shorter landing distances. In this case the requirements of § 29.67(a)(1) may be limiting. The climb performance and climb speeds required by § 29.67(a)(1) and (2) should be established prior to Category A landing tests.

(B) The one-engine-inoperative landing is similar in many respects to the height-velocity tests described in paragraph AC 29.79. Most of the comments, cautions, and techniques for HV also apply here even though typical flight conditions at LDP are less critical than limiting HV points due to a lower power level and an established rate of descent. The approach is made at a predetermined speed and one engine is made inoperative prior to LDP. After the LDP, speed is reduced and the rotorcraft is flared to a conventional one engine inoperative landing. Depending on the landing characteristics and landing profile, the flare may be initiated either prior or subsequent to the 50 foot elevation utilized in determining landing distance. Testing should include an engine failure such that recognition is at the LDP with a 1-second pilot delay to ensure safe landing capability for this critical case. A sufficient number of acceptable runs should be accomplished to provide confidence in the results. Typically ten acceptable runs are adequate.

(C) The balked landing portion of the landing profile is addressed under § 29.85, Balked Landing: Category A. For an explanation of that requirement and a discussion of those test procedures, refer to paragraph AC 29.77.

(iii) Power. Power should be limited to minimum specification values on the operating engine(s). This may be accomplished by adjustment of engine topping to minimum specification values for the range of atmospheric variables to be approved. This is frequently done by installing an adjustable device in the throttle linkage with a control in the cockpit so that engine topping can be accurately adjusted for varying ambient conditions. With such a device in the control system, it becomes vitally important to check topping power prior to each test sequence.

(iv) Aircraft Loading. Aft center of gravity is usually most critical for landing distance determination because visibility constraints limit the degree to which the pilot can flare the rotorcraft for landing. If a weight effect is shown, a minimum of two weights should be flown at each test altitude. One weight should be the maximum weight for prevailing conditions and the other should provide a sufficient spread to validate weight accountability.

(v) Extrapolation. Landing data may be extrapolated along an established W/σ line to the maximum gross weight of the rotorcraft. However, extrapolation will not be considered valid if landing gear loads are marginally acceptable at actual landing weights below the W/σ limit. If no marginal areas are apparent and an acceptable analytical method is used, performance data may be extrapolated up to 4,000 feet density altitude from test conditions. (See paragraph AC 29.45.)

(vi) Ambient Conditions. Appropriate test limits for ambient conditions such as wind and temperature are contained in paragraph AC 29.45. Test data should be corrected for existing wind conditions during landing distance tests. Credit for headwind conditions may be given during flight manual data expansion. Paragraph AC 29.1587 details allowable wind credit.

(vii) All Engine Out Landing.

(A) Several procedures can be utilized to demonstrate compliance with the all-engine-out landing requirement. As discussed in the explanation portion of this paragraph, §§ 29.79 and 29.83 each require that a landing from autorotation be possible. The maneuver is entered by smoothly reducing power at an optimum autorotation airspeed at a safe height above the landing surface. All-engine-out landing tests should be initiated at light weight with a gradual buildup to the limiting weight conditions. If a complete company test program has documented all-engine-out landings to the GW/σ limit, the buildup conditions during verification test may be decreased. If not, buildup testing should be initiated at light weight. This test is ordinarily conducted at mid center of gravity. Typically, all altitudes may be approved with two weight limit landings - one at sea level and one near maximum takeoff and landing altitude.

(B) Demonstrated compliance with this requirement is intended to show that an autorotative descent rate can be arrested, and forward speed at touchdown can be controlled to a reasonable value (less than 40 KTAS is recommended) to ensure a reasonable chance of survivability for the all engine failure condition. On multiengine rotorcraft, rotor inertia is typically lower than for single-engine rotorcraft. RPM decays rapidly when the last engine is made inoperative. Due to this relatively low inertia level, considerable collective may be needed to prevent rotor overspeed conditions when the rotorcraft is flared for landing. Also, when testing the final maximum weight points, the pilot should anticipate a need for considerable collective pitch to control rotor overspeed during autorotative descent, particularly at high altitude WAT limiting conditions. Some designs incorporate features which may lead to rotorcraft damage in testing this requirement (e.g., droop stop breakage or loss of directional control with skids) if landings are conducted to a full stop with the engines cut off.

(C) The intent of this rule is to demonstrate controlled touchdown conditions and freedom from loss of control or apparent hazard to occupants when landing with all engines failed. In these cases compliance can be demonstrated by leaving throttles in the idle position and ensuring no power is delivered to the drive train.

Also, computer analysis may be used in conjunction with simulated in-flight checks to give reasonable assurance that an actual safe touchdown can be accomplished. Another method may be to make a power recovery after flare effectiveness of the rotorcraft has been determined. Other methods may be considered if they lead to reasonable assurance that descent can be arrested and forward speed controlled to allow safe landing with no injury to occupants when landing on a prepared surface with all engines failed. Regardless of the method(s) used to comply with this requirement, careful planning and analyses are very important due to the potentially hazardous aspects of power off simulation and landing of a multiengine rotorcraft totally without power. Considerations for weight and altitude extrapolation are the same as those for HV testing (see paragraph AC 29.79). The all-engine-inoperative landing test is ordinarily done in conjunction with height velocity tests because ground and onboard instrumentation requirements are the same for both tests.

(D) Prior to conducting these tests, the crew should be familiar with the engine inoperative landing characteristics of the rotorcraft. The flight profile may be entered in the same manner as a straight-in practice autorotation. It is recommended that for safety reasons idle power be used if a “needle split” (no engine power to the rotor) can be achieved. In some cases, a low engine idle adjustment has been set to assure needle split is attained. In other cases, a temporary detent between idle and cutoff was used on the throttle. In a third case, the engine was actually shut down on sample runs to verify that the engine power being delivered was not materially influencing landing capability or landing distances. The flare is maintained as long as is reasonable to dissipate speed and build RPM. Rotor RPM should stay with allowable limits. Aft center of gravity is ordinarily critical due to visibility and flare-ability. Following the flare, the rotorcraft is allowed to touch down in a landing attitude. Rotor RPM at touchdown should be recorded, and it should be within allowable structural limits.

(viii) <u>Vertical Landings</u>. The reader should be familiar with the preceding discussion of conventional Category A, landing profiles because duplicate information is not repeated here. A typical vertical landing profile is shown in figure AC 29.75A-2. This profile is equally applicable to both ground level and elevated heliport sites. The profile begins at a stabilized single engine approach condition. It should be possible to make a safe OEI landing or go-around at any point prior to the LDP unless alternate AEO approach procedures are presented in the Flight Manual according to paragraph AC 29.75b(1)(i)(A). It is possible to have two landing techniques: an “offset” one, which schedules drop down for elevated heliports (but still ensure 15 feet radial deck edge clearance), and a “straight in” approach which utilizes the ground level heliport criteria. These techniques should be stipulated as such in the Flight Manual. At the LDP the aircraft becomes committed to landing. A safe landing should be possible in case of an engine failure at any point before or after the LDP. Testing should include a simulated failure at LDP with a 1-second delay or normal pilot response time, whichever is longer, and subsequent landing within the allowable area. The landing distance is the distance from the point at which the lowest portion of the rotorcraft reaches 50 feet above the landing surface to the forward-most point after coming to a stop (including main rotor tip path). The LDP becomes very important for landing on small, elevated heliports. The LDP should be clearly defined and Flight Manual instructions should carefully explain any pilot procedures. An illustration similar to figure AC 29.75A-2 with somewhat more detailed information is most useful. Night OEI landings should be conducted to verify suitable visibility for both internal and external vertical landing cues. The minimum elevated heliport size demonstrated for the OEI approach procedure and for alternate AEO approach procedures (when provided) should also be provided in the Flight Manual.

c. Category B Requirements.

(1) Explanation. Section 29.83 contains the Category B landing requirements. Landing distance is measured from the 50-foot point to the point at which the rotorcraft is completely stopped (approximately 3 knots for water landings). The approach speed is selected by the applicant. Appropriate ambient conditions and allowable extrapolation are discussed under paragraph AC 29.45.

(2) Procedures.

(i) Landing Distance. Aft center of gravity is ordinarily critical due to field-of-view and flare ability. For wheeled rotorcraft, the brakes are applied to an incipient skid for most efficient stopping. For rotorcraft on skids, the collective should be lowered as soon as characteristics allow in order to place a greater weight on the landing skids. These procedures would be appropriate flight manual entries to show how landing distances can be realized. For flight manual purposes, the landing distance should include the horizontal distance from the point at which the lowest part of the rotorcraft first reaches 50 feet above the landing surface to the point at the foremost part of the rotorcraft (including rotor tip path) after coming to a stop. Multiengine rotorcraft incorporating Category A engine isolation features may elect to show compliance with § 29.79 and § 29.81. A sufficient number of acceptable runs should be accomplished to provide confidence in the results. Typically ten acceptable runs are adequate. If a weight effect on landing distance is to be shown, a minimum of two weight extremes are normally tested.

(ii) All-Engine-Out Landing.

(A) Several procedures can be utilized to demonstrate compliance with the all-engine-out landing requirement. Section 29.83(c) requires that a landing from autorotation be possible. The maneuver is entered by smoothly reducing power at an optimum autorotation airspeed at a safe height above the landing surface. All-engine-out landing tests should be initiated at light weight with a gradual buildup to the limiting weight conditions. If a complete company test program has documented all-engine-out landings to the GW/σ limit, the buildup conditions during verification test may be decreased. This test is ordinarily conducted at mid center of gravity. Typically, all altitudes may be approved with two weight limit landings - one at sea level and one near maximum takeoff and landing altitude.

(B) Demonstrated compliance with this requirement is intended to show that an autorotative descent rate can be arrested, and forward speed at touchdown can be controlled to a reasonable value (less than 40 KTAS is recommended) to ensure a reasonable chance of survivability for the all engine failure condition. On multiengine rotorcraft, rotor inertia is typically lower than for single-engine rotorcraft. RPM decays rapidly when the last engine is made inoperative. Due to low rotor inertia, considerable collective may be needed to prevent rotor overspeed conditions when the rotorcraft is flared for landing. Also, when testing the final maximum weight points, the pilot should anticipate a need for considerable collective pitch to control rotor overspeed during autorotative descent, particularly at high altitude WAT limiting conditions.

(C) The intent of this rule is to demonstrate controlled touchdown conditions and freedom from loss of control or apparent hazard to occupants when landing with all engines failed. In these cases compliance can be demonstrated by leaving throttles in the idle position and ensuring no power is delivered to the drive train. Also, computer analysis may be used in conjunction with simulated in-flight checks to give reasonable assurance that an actual safe touchdown can be accomplished. Another method may be to make a power recovery after flare effectiveness of the rotorcraft has been determined. Other methods may be considered if they lead to reasonable assurance that descent can be arrested and forward speed controlled to allow safe landing with no injury to occupants when landing on a prepared surface with all engines failed. Regardless of the method(s) used to comply with this requirement, careful planning and analyses are very important due to the potentially hazardous aspects of power off simulation and landing of a multiengine rotorcraft totally without power. Considerations for weight and altitude extrapolation are the same as those for HV testing (see paragraph AC 29.79). The all-engine-inoperative landing test is ordinarily done in conjunction with height velocity tests because ground and onboard instrumentation requirements are the same for both tests.

(D) Prior to conducting these tests, the crew should be familiar with the engine inoperative landing characteristics of the rotorcraft. The flight profile may be entered in the same manner as a straight-in practice autorotation. It is recommended that for safety reasons idle power be used if a “needle split” (no engine power to the rotor) can be achieved. In some cases, a low engine idle adjustment has been set to assure needle split is attained. In other cases, a temporary detent between idle and cutoff was used on the throttle. In a third case, the engine was actually shut down on sample runs to verify that the engine power being delivered as not materially influencing landing capability or landing distances. The flare is maintained as long as is reasonable to dissipate speed and build RPM. Rotor RPM should stay with allowable limits. Aft center of gravity is ordinarily critical due to visibility and flareability. Following the flare, the rotorcraft is allowed to touch down in a landing attitude. Rotor RPM at touchdown should be recorded, and it should be within allowable structural limits.

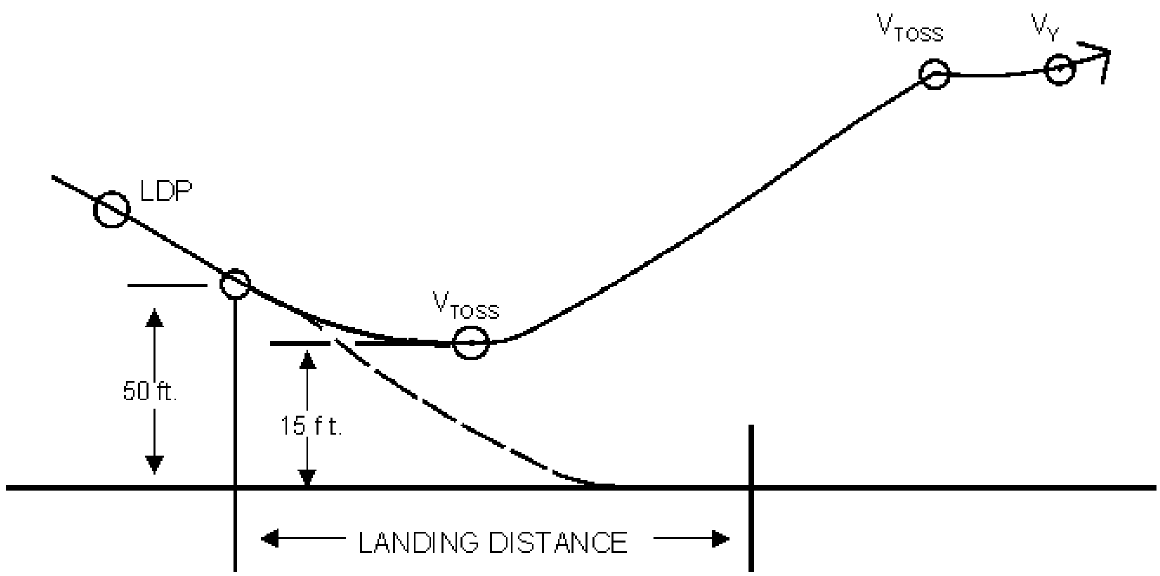


FIGURE AC 29.75A-1 CATEGORY A CONVENTIONAL LANDING - CLEAR HELIPORT

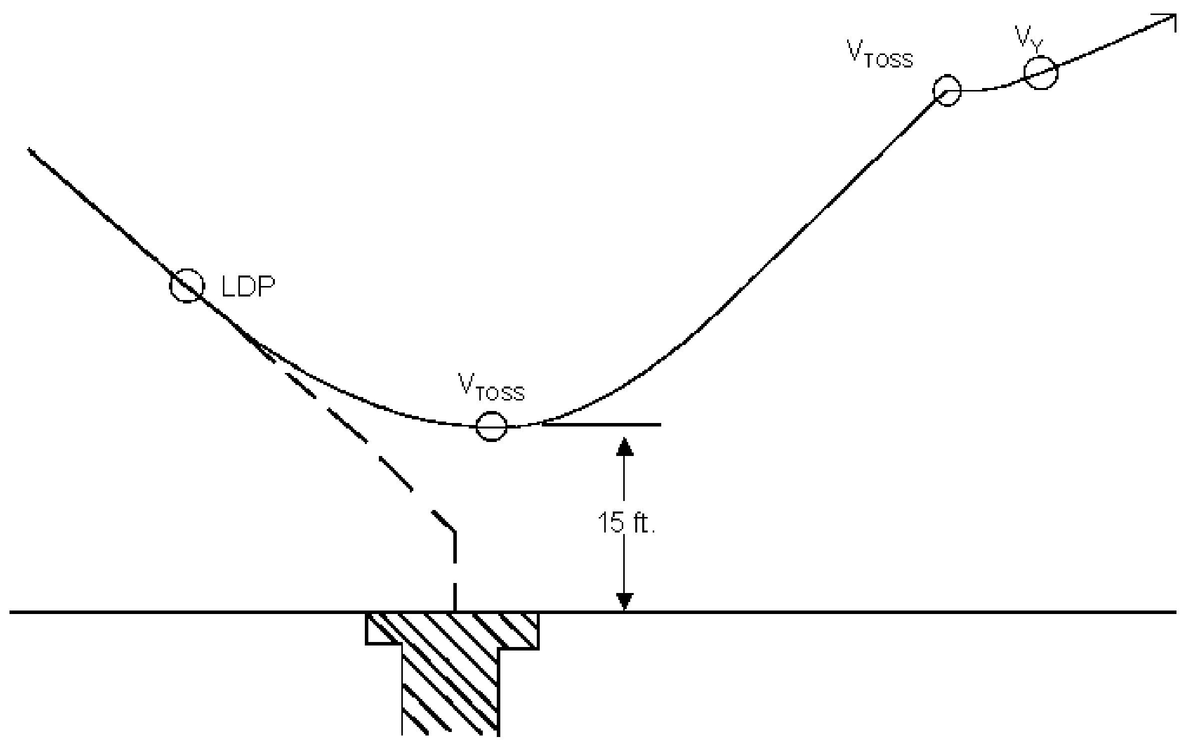


FIGURE AC 29.75A-2 CATEGORY A VERTICAL LANDING

AC 29.77. § 29.77 (Amendment 29-24) BALKED LANDING: CATEGORY A

(For § 29.77 after Amendment 38, see paragraph AC 29.75A)

a. Explanation. This rule has two distinct portions.

(1) Section 29.77(a) states that the rotorcraft must be capable of transitioning smoothly from each approved Category A approach condition to a missed approach with one engine inoperative (OEI). Although not specifically stated in the rule, this requirement must be met for any point prior to the landing decision point (LDP).

(2) Section 29.77(b) requires that the LDP be defined so that it will permit transition to a safe climb condition in the event a balked landing is necessary. (See figure AC 29.75-1.) The safe climb conditions are defined in § 29.67(a)(1) and (2). This suggests establishing a clearly defined balked landing profile similar to the Category A takeoff profile established under § 29.59. The balked landing profile must insure compliance with the climb performance requirements of §§ 29.67(a)(1) and 29.67(a)(2).

b. Procedures.

(1) Instrumentation. Instrumentation requirements are similar to those for Category A takeoff. A ground station with positioning capability is needed along with on-board instrumentation of engine and flight parameters.

(2) Balked Landing Profiles. One engine inoperative balked landing profiles during approach must be conducted at conditions up to and including the LDP. The LDP should be designated so that the balked landing profile may be completed with the rotorcraft descending no lower than 35 feet above the landing surface. The distance from the LDP to the point in the balked landing profile at which a minimum of 35 feet above the landing surface is attained at V_{TOSS} in a climbing posture should be recorded. This distance should be compared against the landing distance determined under § 29.81 to assure the balked landing maneuver can be completed within the designated landing area. This is especially important for future steep angle, low speed Category A approaches to heliports.

(3) Handling Qualities. Handling qualities features in the balked landing transition should be carefully evaluated. Characteristics such as excessive nose down pitching with power application or excessive engine lag should not be approved.

(4) Climb Performance. In accordance with this rule, the climb requirements of § 29.67(a)(1) and (2) must also be met in the event a balked landing is made. See paragraphs AC 29.65 and AC 29.67.

AC 29.79. § 29.79 (Amendment 29-21) LIMITING HEIGHT-SPEED ENVELOPE.

(For § 29.79 after Amendment 29-38, see paragraph AC 29.75A)

a. Explanation.

(1) The height-speed envelope is normally referred to as the height-velocity (HV) diagram. It defines an envelope of airspeed and height above the ground from which a safe power-off or OEI landing cannot be made. The diagram normally consists of three portions: (a) the level flight (cruise) portion, (b) the takeoff portion, and (c) the high speed portion. See figure AC 29.79-1. The high speed portion is omitted on occasions when it can be shown that the rotorcraft can suffer an engine failure at low altitude and high speed (up to V_H) and make a successful landing, or climb out on the remaining engine(s).

(2) Engine power considerations are similar to those in previous takeoff and landing requirements, paragraphs AC 29.53, AC 29.63, and AC 29.75.

(3) The prohibited sections of the HV diagram are separated by the takeoff corridor. This corridor should be wide enough to consistently permit a takeoff flight path clear of the HV diagram using normal pilot skill. The takeoff corridor should always permit a minimum of $\pm$ 5 knots clearance from critical portions of the diagram.

(4) The knee of the curve separates the takeoff portion from the cruise portion and is defined as the highest speed point on the low speed portion of the HV envelope. Altitudes above this point are considered cruise, or “fly-in,” points and these test points require a minimum time delay of 1-second between throttle chop and control actuation (reference § 29.143(d)). Altitudes below the knee represent takeoff profile points. For test points in the takeoff portion, takeoff power (or a lower power selected by the applicant as an operating limitation), and normal pilot reaction time will be used.

(5) Since the HV diagram may represent the limiting capabilities of the rotorcraft, each test point should be approached with caution. The manufacturer’s buildup program should be reviewed to determine the amount of conservatism in the HV diagram (if any). It should be remembered that the operational pilot will be operating at or near the HV diagram without the benefit of a buildup program. Buildup testing is necessary, and it is most important to vary only one parameter at a time to prevent surprises. Light weight testing is ordinarily conducted first. High and low hover points are approached from above and below respectively. Portions near the knee are initially evaluated at high speed with subsequent backing down of the speed. In most rotorcraft the effective flare airspeed is critical. At airspeeds slightly below this value, the ability to arrest and control descent rates through use of an aft cyclic flare may be greatly diminished. Extreme care should be exercised when “backing down” to lower speeds.

(6) In addition to the on-board and ground instrumentation, a motion picture camera or other position measuring equipment should cover each run.

(7) For FAA/AUTHORITY tests, the minimum required crew and minimum instrument panel display should be used. Ground safety equipment should be provided.

(8) This test is the least predictable of all the performance items. Therefore, the expansion and extrapolation of test data are questionable. Weight may not be extrapolated to higher values. In order to extrapolate HV data to higher altitudes, any analytical method must have FAA/AUTHORITY approval. In lieu of pure analytical methods, simulations have been used successfully, especially for multiengine rotorcraft. In either case, the maximum allowable extrapolation should be limited to 2,000 feet density altitude (H_D). HV test weights should be consistent with the takeoff and landing WAT (weight, altitude, temperature) limit curve which will be placed in the Rotorcraft Flight Manual. For a given diagram, typical weight reductions that are necessary as altitude is increased can be conservatively estimated by maintaining a constant gross weight divided by density ratio, GW/σ. See figure AC 29.79-2, Part A. If weight is not varied, an enlarged HV diagram is required for safe power-off landing as density altitude is increased. See figure AC 29.79-2, Part B. Another method of presentation is to show varying weights at a constant density altitude. (See figure AC 29.79-2, Part C.)

(9) Vertical takeoff and landing (VTOL) testing normally does not require separate HV testing. The takeoff and landing tests take on the combined characteristics of takeoff, landing, and HV tests.

(10) Rotorcraft certificated prior to Amendment 29-21 were required to have the resulting height-velocity diagram as an operating limitation. This limitation restricted opportunities when operating large rotorcraft in various utility applications. Subsequently, Amendment 29-21 allows, under certain conditions, the height-velocity diagram to be placed in the Flight Manual Performance Information Section instead of the Limitations Section. Specifically, the rotorcraft must be: (1) certificated for a maximum gross weight of 20,000 pounds or less; (2) configured with nine passenger seats or less; and (3) certificated in Category B. Testing must be completed with the aircraft at the maximum gross weight at sea level. For altitudes above sea level, the test aircraft must be at a weight no less than the highest weight the rotorcraft can hover out-of-ground-effect (OGE). Rotorcraft certificated prior to Amendment 29-21 can update their certification basis to take advantage of this provision.

b. Procedures.

(1) Instrumentation.

(i) Ground Station. The ground station must have equipment and instrumentation to determine wind direction and velocity, outside air temperature, and (if the test rotorcraft has reciprocating engines), humidity. Since the tests must be conducted in winds of 2 knots or less, a smoke generator is highly recommended to show both flightcrew and ground crew personnel the wind direction and velocity at any given time. Additionally, the location of the ground station should be such that it is free of rotor downwash at all times. Motion picture, phototheodolite, and radio equipment will be necessary to properly conduct the test program. The use of telemetry equipment is desirable if the location of the test site and the magnitude of the test program make it practical.

(ii) Airborne Equipment (Test Rotorcraft). Necessary installed test equipment may include photo panels and/or recorders for recording engine parameters, control positions, landing gear loads, landing gear deflections, airspeed, altitude, and other variables. An external light attached to the rotorcraft (or any other means of identifying the engine failure point to the ground camera or phototheodolite) is needed to identify the exact time of engine failure and may also be used to synchronize the ground recorder with the airborne recorded data.

(2) Analytical Prediction. The HV diagram can be estimated by analytical means and this is recommended prior to test. HV, however, is the least predictable of all rotorcraft performance and because of this, the expansion and extrapolation of test data must be done with great care. Test weight may not be extrapolated. All test points should be approached conservatively with some speed or altitude margin. If the manufacturer has conducted a comprehensive HV flight test program to validate his analytical predictions, much preliminary testing can be eliminated. In any case, the maximum allowable extrapolation from flight test conditions is 2,000 feet density altitude and an approved analytical and/or simulation method must be utilized for extrapolation.

(3) Power.

(i) The appropriate power level before engine failure for the low and high hover points is simply the power required to hover at the prevailing hover conditions. The appropriate power condition prior to failure of the engine for points below the knee is takeoff power or a lower value if approved as an operating limit. For cruise or “fly-in” points above the knee, the appropriate condition is power required for level flight. Rotor speed at execution of the engine failure should be the minimum speed appropriate to the flight condition.

(ii) The applicable power failure conditions are listed in § 29.79(b). Power should be completely cut for normal Category B rotorcraft. For Category A rotorcraft, the desired topping power (for the remaining engine(s)) should be set prior to the test. This power value will need adjustment as ambient conditions change. The power can be takeoff power (TOP), 2 1/2-minute power, or some calculated lower power for simulating hot day or higher density altitude conditions. Power is verified and recorded by the pilot by “topping” the engine(s) prior to engine failure tests. Care must be taken to assure that this power value is no more than that which would be delivered by a minimum specification engine under the ambient conditions to be approved.

(4) Test Loadings. Weight extrapolation is not permitted for HV. Therefore, the test weight must be closely controlled. Ballast or fuel should be added frequently to maintain the weight within -1 to +5 percent when testing final points. Ordinarily tests are

conducted at a mid center of gravity unless a particular loading is expected to be particularly critical.

(5) Landing Gear Loads.

(i) Instrumented landing gear can be a great help in evaluating test results. This information can be telemetered to a ground station or otherwise recorded and displayed for direct reference following each landing.

(ii) Any landing which results in permanent deformation of aircraft structure or landing gear beyond allowable maintenance limits is considered an unsatisfactory test point.

(6) Piloting Considerations. In verifying the HV diagram, the minimum required instrument panel display and minimum crew should be used in order not to mislead the operational pilot who has no test equipment available and may have no copilot to assist. Three distinctly different flight profiles are utilized in developing the diagram.

(i) High Hover. A stabilized out-of-ground-effect (OGE) hover condition prior to power failure is essential. A minimum 1-second time delay between power failure and initial control actuation is utilized. Following the time delay, the primary concern is to quickly lower collective and to gain sufficient airspeed to allow an effective flare approaching touchdown. While the immediate development of airspeed is necessary, the dive angle must be reasonable and must be representative of that expected in service. While initial aircraft attitude will vary between models and with changing conditions, 10°-20° has been previously applied as a maximum allowable nose down pitch attitude. Use of greater attitudes could result in a diagram which is difficult to achieve and unrealistic for operations in service. Initial testing should start relatively high with gradual lowering of height to the final high hover altitude. A stabilized OGE hover condition prior to power failure is essential. If a stabilized high hover condition cannot be achieved prior to the engine cut, then this point should be tested from a minimum level flight speed. This will result in an open-ended HV diagram. A smoke source or balloon on a long cord is highly desirable since the wind can vary significantly from surface observations to typical high hover altitudes. Vertical speed must be very near zero at the throttle chop. Any climb or sink rate can have a significant influence on the success of the test point. Use of a radar altimeter with a cross check to barometric altitude is essential.

(ii) Low Hover. From the low hover position there is no flare capability and little time for collective reaction. No time delay is applied other than normal pilot reaction. For typical designs the collective may not be lowered after power failure. Lowering of the collective is not permitted because it is not a pilot action which could be expected if an engine failed without notice during a hovering condition in service. Initial lowering of collective immediately after power failure can result in very high, unconservative low hover altitudes that are unrealistic for operational conditions. If, however, a design is such that a 1-second pilot delay after power failure could be achieved without any appreciable descent, a slight lowering of collective could be allowed.

(iii) Takeoff Corridor. Normal pilot reaction is applied when the engine is made inoperative. At low speeds collective may be lowered quickly to retain RPM and minimize the time between power failure and ground contact. If airspeed is sufficient for an effective flare, the aircraft is flared to reduce airspeed, retain rotor RPM, and control vertical speed prior to touchdown. Considerable surface area may be needed for a sliding or rolling stop.

(iv) Additional Considerations. The “in-between” points utilize similar techniques. The cruise or “fly-in” points are similar to the high hover point although the steep initial pitch attitudes are not needed as altitude is decreased and airspeed is increased along the curve. The low speed points along the takeoff corridor are similar to the low hover point except that the collective may be quickly lowered and some flare capability may be used as the “knee” is approached. The pilot should be proficient in all normal autorotation landings before conducting HV tests in a single-engine rotorcraft.

(7) Ground Support. Motion picture or theodolite coverage and ground safety equipment are necessary. Communication capability among these elements should be provided. Use of a phototheodolite to compare height/speed with cockpit observations is very desirable.

(8) Verifying the HV Diagram.

(i) A sufficient number of test points must be flown to verify the diagram. The key areas are the knee, high altitude hover, low altitude hover, and high speed touchdown. Test points with excessive gear loads, above average skill requirements, winds above permissible levels, rotor droop below approved minimum transient RPM, damage to the rotorcraft, excessive power, incorrect time delay, etc., cannot be accepted.

(ii) After the HV diagram is defined, it should be ascertained that the corridor permits takeoffs within ±5 knots of the recommended takeoff profile.

(9) Flight Manual. The flight manual should list any procedures which may apply to specific points (e.g., high speed points) and test conditions, such as runway surface, wave height for amphibious tests, marginal areas of controllability or landing gear response, etc. The HV curve should be presented in the RFM using actual altitude above ground level and indicated airspeed.

(10) Night Evaluation. If a rotorcraft is to be certified for night operation, a night evaluation is required. Engine failures should be conducted along the recommended takeoff path. Landings should also be qualitatively evaluated with an engine failed. Engine failures at critical HV conditions are not required. The intent is to show

adequate visibility using aircraft and/or runway lights without requiring a duplication of the daytime HV test program. See related discussion under paragraph AC 29.63.

(11) Water Landings. For amphibious float equipped rotorcraft, day and night water landings should be conducted under critical loading conditions with an engine failed. Engine failures should be conducted along the recommended takeoff path. Engine failures at critical HV conditions are not required. The intent is to show similarity to test results over land without requiring a duplication of the HV test program.

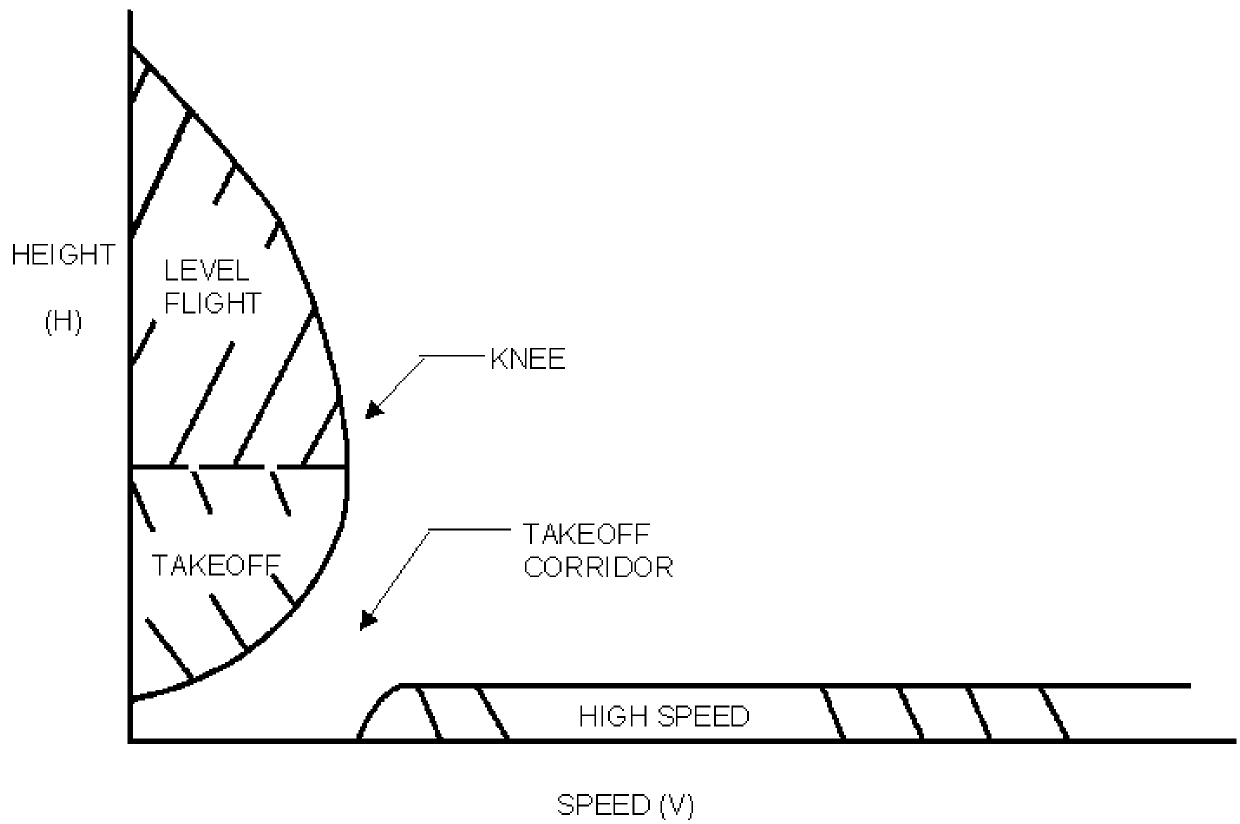


FIGURE AC 29.79-1 HEIGHT - VELOCITY DIAGRAM

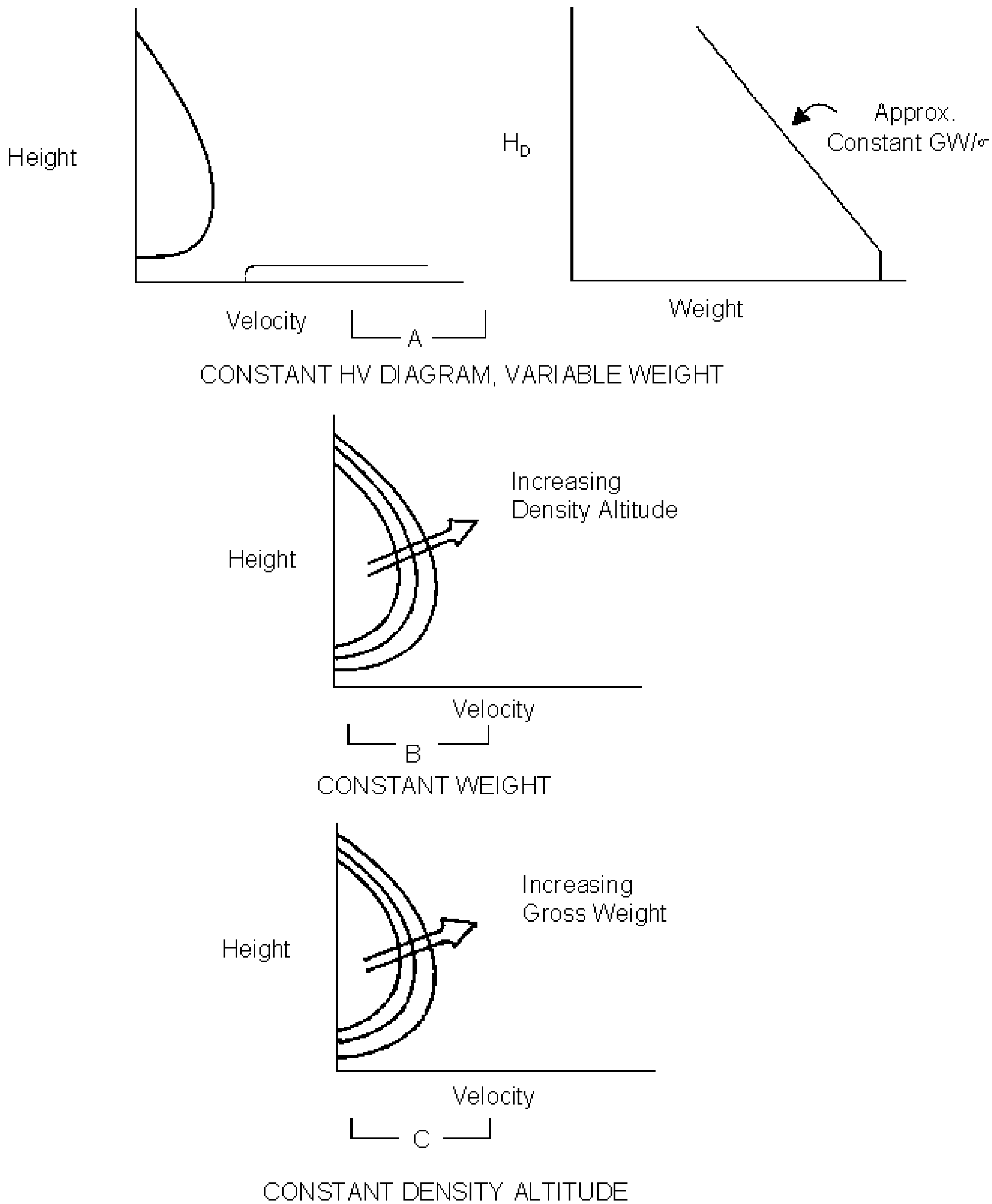


FIGURE AC 29.79-2 ALTITUDE/WEIGHT ACCOUNTABILITY

AC 29.85. <u>§ 29.85 (Amendment 29-39) BALKED LANDING: CATEGORY A</u>

(For Balked Landing prior to Amendment 39, see § 29.77 and paragraph AC 29.77.)

a. <u>Explanation</u>. Amendment 29-39 revised and relocated the original § 29.77 as a new § 29.85. The guidance material of paragraph AC 29.77 does not apply to rotorcraft certified with Amendment 29-39 or later. This rule has two distinct portions.

(1) Section 29.85(a) states that the rotorcraft must be capable of transitioning smoothly from each approved Category A approach condition to a missed approach with one engine inoperative (OEI). Although not specifically stated in the rule, this requirement must be met for any point prior to the landing decision point (LDP).

(2) Section 29.85(b) requires that the LDP be defined so that it will permit transition to a safe climb condition in the event a balked landing is necessary. (See figure AC 29.75A-1.) The safe climb conditions are defined in § 29.67(a)(1) and (2). A clearly defined balked landing profile similar to the Category A takeoff profile should be established. The balked landing profile must insure compliance with the climb performance requirements of §§ 29.67(a)(1) and 29.67(a)(2).

b. <u>Procedures</u>.

(1) <u>Instrumentation</u>. Instrumentation requirements are similar to those for Category A takeoff. A ground station with positioning capability is needed along with on-board instrumentation of engine and flight parameters.

(2) <u>Balked Landing Profiles</u>. One engine inoperative balked landing profiles during approach must be conducted at conditions down to and including the LDP. The LDP should be designated so that the balked landing profile may be completed with the rotorcraft clearing the landing surface by a minimum of 15 feet. Fifteen feet should be considered the absolute minimum clearance allowed with greater clearances required for some rotorcraft dependent on rotorcraft geometry and performance characteristics. For elevated or ground level heliports, with significantly lower LDP heights than 100 feet, the minimum clearance is 15 feet vertically and radially. These minimum heights would need to be demonstrated with variations in piloting techniques and with pilot recognition and reaction times for engine failures occurring before/after LDP. The distance from the LDP to the point in the balked landing profile at which a minimum of 35 feet above the landing surface is attained at V_{TOSS} in a climbing posture should be recorded. This distance should be compared against the landing distance determined under § 29.81 to assure the balked landing maneuver can be completed within the designated landing area. This is especially important for future steep angle, low speed Category A approaches to heliports.

(3) <u>Handling Qualities</u>. Handling qualities features in the balked landing transition should be carefully evaluated. Characteristics such as excessive nose down pitching with power application or excessive engine lag should not be approved.

(4) Climb Performance. In accordance with this rule, the climb requirements of § 29.67(a)(1) and (2) must also be met in the event a balked landing is made. See paragraphs AC 29.65 and AC 29.67.

AC 29.87. § 29.87 (Amendment 29-39) LIMITING HEIGHT-SPEED ENVELOPE.

(For Limiting Height-Speed Envelope prior to Amendment 39, see § 29.79 and paragraph AC 29.79.)

a. Explanation. Amendment 39 redesignated § 29.79 as § 29.87.

b. Procedures. The guidance material presented in paragraph AC 29.79 continues to apply.

SUBPART B - FLIGHT

FLIGHT CHARACTERISTICS

AC 29.141. § 29.141 (Amendment 29-24) FLIGHT CHARACTERISTICS - GENERAL.

a. Explanation.

(1) This regulation prescribes the general flight characteristics required for certification of a transport category rotorcraft. Specifically, it states that the rotorcraft shall comply with the flight characteristics requirements at all approved operating altitudes, gross weights, center of gravity locations, airspeeds, power, and rotor speed conditions for which certification is requested. The reference to "altitude" in § 29.141(a)(1) refers to "density altitude." Density altitude is, of course, a function of pressure altitude and ambient temperature, hence the need to account for ambient temperature effects. Additional flight characteristics required for instrument flight are contained in AC 29 Appendix B.

(2) Generally the aircraft structural (load level) survey accounts for takeoff power values at speeds up to and including V_Y. At speeds above V_Y, maximum continuous power is assumed. Stress to rotating components usually increases with airspeed and power. If the takeoff power rating exceeds the maximum continuous power rating, and the structural survey has been conducted under the assumption that takeoff power is not used at speeds above V_Y, the Rotorcraft Flight Manual must limit takeoff power to speeds of V_Y and below. If takeoff power is structurally substantiated throughout the flight envelope, and appropriate portions of the controllability, maneuverability, and trim requirements of §§ 29.141 through 29.161 are met at takeoff power levels, no flight manual entry is needed. Obviously, if transmission limits for MC and takeoff power are coincident, no special action is needed.

(3) During the flight characteristics testing, the controls must be rigged in accordance with the approved rigging instructions and tolerances. The control system rigging must be known prior to testing. In addition to the normal rigging procedures, any programmed control surfaces which may be operated by dynamic pressure, electronics, etc., must also be calibrated. During the flight test program, it is frequently necessary to rig a control, such as the swashplate or tail rotor blade angle, to the allowable critical extreme of the tolerance band. For example, it would be necessary to rig the tail rotor to the minimum allowable blade angle if meeting the requirements of §29.143(c) would be in question. The same consideration must be given to all rotorcraft controls and moveable aerodynamic surfaces where questionable compliance with the regulations may exist. If the rotor-induced vibration characteristics of the rotorcraft are significantly affected and require time-consuming rigging for such things as acceptable ride comfort, then the rotor(s) should be rigged to the allowable extreme tolerance limits to determine compliance, for example, with § 29.251.

(4) During the FAA/AUTHORITY flight test program, the crew should be especially alert for conditions requiring great attentiveness, high skill levels, or exceptional strength. If any of these features appear marginal, it is advisable to obtain another pilot's opinion and to carefully document the results of these evaluations. Section 29.141(b) provides the regulatory basis for these strength and skill requirements. The general requirements for a smooth transition capability between appropriate flight conditions are also included in § 29.141(b). These requirements must also be met during appropriate engine failure conditions for each category of rotorcraft.

(5) For night or IFR approval, § 29.141(c) contains the general regulatory reference which requires additional characteristics for night and IFR flight. The appropriate flight test procedures are included in other portions of this order.

AC 29.143. § 29.143 (Amendment 29-24) CONTROLLABILITY AND MANEUVERABILITY.

a. Explanation.

(1) This regulation contains the basic controllability requirements for transport rotorcraft. It also specifies a minimum maneuvering capability for required conditions of flight. The general requirements for control and for maneuverability are summarized in § 29.143(a) which is largely self-explanatory. The hover condition is not specifically addressed in § 29.143(a)(2) so that the general requirement may remain applicable to all rotorcraft types, including those without hover capability. For rotorcraft, the hover condition clearly applies under "any maneuver appropriate to the type."

(2) Paragraphs (b) through (e), § 29.143, include more specific flight conditions and highlight the typical areas of concern during a flight test program.

(i) Section 29.143(b) specifies flight at V_{NE} with critical weight, center of gravity (CG), rotor RPM, and power. Adequate cyclic authority must remain at V_{NE} for nose down pitching of the rotorcraft and for adequate roll control. Nose down pitching capability is needed for control of gust response and to allow necessary flight path changes in a nosedown direction. Roll control is needed for gust response and for normal maneuvering of the aircraft. In the past, 10 percent control margin has been applied as an appropriate minimum control standard. The required amount of control power, however, has very little to do with any fixed percentage of remaining control travel. There are foreseeable designs for which 5 percent remaining is adequate and others for which 20 percent may not be enough. The key is, can the remaining longitudinal control travel at V_{NE} generate a clearly positive nose down pitching moment, and will the remaining lateral travel allow at least 30° banked turns at reasonable roll rates? Moderate lateral control reversals should be included in this evaluation and since available roll control can diminish with sideslip, reasonable out of trim conditions (directionally) should be investigated. This "control remaining" philosophy must also be applied for other flight conditions specified in this section.

(ii) Section 29.143(c) requires a minimum 17-knot control capability for hover and takeoff in winds from any azimuth. Control capability in wind from zero to at least 17 knots must also be shown for any other appropriate maneuver near the ground such as rolling takeoffs for wheeled rotorcraft. These requirements must be met at all altitudes approved for takeoff and landing. On rotorcraft incorporating a tail rotor, efficiency of the tail rotor decreases with altitude so that a given sideward flight condition requires more pedal deflection, a higher tail rotor blade angle, and more horsepower. Hence, directional capability in sideward flight (or at critical wind azimuth) is most critical during testing at a high altitude site. Prior to Amendment 29-24, hover controllability, height-velocity, and hover performance were the three regulatory requirements that ordinarily determined the shape of the limiting weight-altitude-temperature (WAT) curve for takeoff and landing. For Category A performance rotorcraft operations, of course, the one-engine-inoperative climb performance requirements may also influence the WAT limit curve. Amendment 29-24 allows, under certain conditions, the deletion of any hover controllability condition determined under Section 29.143(c) from becoming an operating limitation. Section 29.1587 of Amendment 29-24 provides a means wherein Category B certificated rotorcraft (in accordance with the requirements of 29.1, effective with Amendment 29-21) may not be limited by the hover controllability requirements of 29.143(c). Section 29.1583(g) requirements for Category A certificated rotorcraft are unchanged from past regulatory requirements in that if the hover controllability requirements of 29.143(c) result in the most restrictive envelope it will be published as an operating limitation. Section 29.1587(b) provides a means wherein Category B certificated rotorcraft, as defined in FAR 29.1, may not be restricted in its utilization. It allows such rotorcraft to publish the maximum takeoff and landing capabilities of the rotorcraft, provided something other than the 17 knot hover controllability requirement is not limiting. This may be zero wind IGE hover performance or any other performance the applicant elects to use if the maximum safe wind for operations near the ground is provided. Rotorcraft certificated prior to Amendment 29-24 can update their certification basis to take advantage of this provision. If an applicant with a previously type certificated rotorcraft elects to update to this later amendment, caution should be taken to verify that the height-velocity information is done in accordance with Amendment 29-21; that all engine out landing capabilities are satisfactorily accounted for at the new proposed gross weight, altitude, temperature combinations; that takeoff/landing information is provided; and that sufficient information is provided to properly advise the crew of the rotorcraft's capabilities when utilizing this increased performance capabilities.

(iii) Section 29.143(d) requires adequate controllability when an engine fails. This requirement specifies conditions under which engine failure testing must be conducted and includes minimum required delay times.

(A) For rotorcraft which meet the engine isolation requirements of Category A, demonstration of sudden complete single-engine failure is required at critical conditions throughout the flight envelope including hover, takeoff, climb at V_Y, and high speed flight up to V_{NE}. Entry conditions for the first engine failure are engine or transmission limiting maximum continuous power (or takeoff power where appropriate) including reasonable engine torque splits. For multiengine Category A installations (three or more engines) subsequent engine failures should be conducted utilizing the same criteria as that used for first-engine failure. The applicant may limit his flight envelope for subsequent failures. Initial or sequential engine failure tests are ordinarily much less severe than the “last” engine failure test required by § 29.75(b)(5). The conditions for last-engine failure are maximum continuous power, or 30-minute power if that rating is approved, level flight, and sudden engine failure with the same pilot delay of 1 second or normal pilot reaction time, whichever is greater.

(B) For Category B powerplant installation rotorcraft, demonstration of sudden complete power failure is required at critical conditions throughout the flight envelope. This includes speeds from zero to V_{NE} (power-on) and conditions of hover, takeoff and climb at V_Y. Maximum continuous power is specified prior to the failure for the cruise condition. Power levels appropriate to the maneuver should be used for other conditions. The corrective action time delay for the cruise failure should be 1 second or normal pilot reaction time (whichever is greater). Cyclic and directional control motions which are part of the pilot task of flight path control are normally not subject to the 1-second restriction; however, the delay is always applied to the collective control for the cruise failure. If the aircraft flying qualities and cyclic trim configuration would encourage routine release of the cyclic control to complete other cockpit tasks during cruise flight, consideration should be given to also holding cyclic fixed for the 1-second delay. Although the same philosophy could be extended to the directional controls, the likelihood of the pilot having his feet away from the pedals is much lower, unless the aircraft has a heading hold feature. Rotor speed at execution of the cruise condition power failure should be the minimum power-on value. The term “cruise” also includes cruise climb and cruise descent conditions. Normal pilot reaction times are used elsewhere. Although this requirement specifies maximum continuous (MC) power, it does not limit engine failure testing to MC power. If a takeoff power rating is authorized for hover or takeoff, engine failure testing must also be accomplished for those conditions in order to comply with § 29.63(c). Following power failure, rotor speed, flapping, and aircraft dynamic characteristics must stay within structurally approved limits.

(iv) Section 29.143(e) addresses the special case in which a V_{NE} (power-off) is established at an airspeed value less than V_{NE} (power-on). For this case, engine failure tests are still required at speeds up to and including V_{NE} (power-on), and the rotorcraft must be capable of being slowed to V_{NE} (power-off) in a controlled manner with normal pilot reactions and skill. There is, however, no controllability requirement for stabilized power-off flight at speeds above 1.1 V_{NE} (power-off) when V_{NE} (power-off) is established per § 29.1505(c).

(v) Application of the controllability requirement for pitch, roll, and yaw at speeds of 1.1 V_{NE} (power-off) and below is similar to that described above for power-on testing at V_{NE}. Sufficient directional control must exist to allow straight flight in autorotation during all approved maneuvers including 30° banked turns up to V_{NE}

(power-off) with some small additional allowance for gust control. Adequate controllability margins must exist in all axes throughout the approved autorotative flight envelope. Testing to V_{NE} at MC power per § 29.143(b), 1.1 V_{NE} at power for 0.9 V_H per § 29.175(b) or § 29.1505, and to 1.1 V_{NE} (power-off) in autorotation per § 29.143(e) should be sufficient to assure adequate control margin during a descent condition at high speed and low power. The high speed, power-on descent condition should be checked for adequate control margin as a “maneuver appropriate to the type.” There has been one instance where insufficient directional pedal was available to maintain a reasonable trimmed sideslip angle with low power at very high speeds, and a case where there was insufficient forward and lateral cyclic available to reach the power-on V_{NE}. The insufficient directional pedal margin was due to the offset vertical stabilizers. The lack of cyclic stick margin was because the cyclic stick migrated to the right as power was reduced and the control limits were circular. This provided less total available forward cyclic stick travel when the cyclic was moved right and forward about 45° from the center position. Each of the above rotorcraft was certificated with a rate of descent limitation to preclude operation in the control-limited area.

(vi) An evaluation of the emergency descent capability of the rotorcraft should be made, either analytically or through flight test. Areas of consideration are the rate of descent available, the maximum approved altitude, and the time before a catastrophic failure following the loss of transmission oil pressure or other similar failure. Each rotorcraft should have the capability to descend to sea level and land from the maximum certificated altitude within the time period established as safe following a critical failure. If the time period does not permit a sea level landing, the maximum height above the terrain must be specified in the limitation section of the Rotorcraft Flight Manual.

(3) The required controllability and maneuvering capabilities must also be considered following the failure of automatic equipment used in the control system (§ 29.672). Examples include stability augmentation systems (SAS), stability and control augmentation systems (SCAS), automatic flight control systems (AFCS), devices to provide or improve longitudinal static stability such as a pitch bias actuator (PBA), yaw dampers, and fly-by-wire elevator or stabilator surfaces. These systems all use actuators of some type, and they are subject to actuator softover and hardover malfunctions. The flight control system should be evaluated to determine whether an actuator jammed in an extreme position would result in reduced control margins. Generally, if the flight control system stops are between the actuator and the cockpit control, the control margin will be affected. If the control stops are between the actuator and the rotor head, the control margins may not be affected, but the location of the cockpit control may be shifted. This could produce interference with other items in the cockpit. An example of this would be a lateral actuator jammed hardover causing a leftward shift in the cyclic stick position. Interference between the cyclic stick, the pilot’s leg, and the collective pitch control could reduce the left lateral control available and reduce left sideward flight capability. In the case of fly-by-wire surfaces, both the high speed forward flight controllability and the rearward flight capabilities could be affected. Flight control systems that incorporate automatic devices should be thoroughly

evaluated for critical areas. Every failure condition that is questionable should be flight tested with the appropriate actuator fixed in the critical failure position. These failures may require limitations of the flight envelope. Any procedure or limitation that must be observed to compensate for an actuator hardover and/or softover malfunction should be included in the Rotorcraft Flight Manual.

b. Procedures.

(1) Flight test instrumentation should include ambient parameters, all flight control positions, rotor RPM, main and tail rotor flapping (if appropriate), engine power instruments, and throttle position. Flight controls that are projected to be near their limits of authority should be rigged to the most adverse production tolerance. A very accurate weight and balance computation is needed along with a precise knowledge of the aircraft's weight/CG variation as fuel is burned.

(2) The critical condition for V_{NE} controllability testing is ordinarily aft CG, MC power, and minimum power-on rotor RPM, although power and RPM variations should be specifically evaluated to verify their effects. The turbine engine is sensitive to ambient temperatures which affect the engine's ability to produce rated maximum continuous torque. Flight tests conducted at ambient temperatures that cause the turbine temperature to limit maximum continuous power would not produce the same results obtained at the same density altitude at colder ambient temperatures where maximum continuous torque would be limiting. Forward CG should be spot checked for any "tuck under" tendency at high speed. The V_{NE} controllability test is normally accomplished shortly after the 1.1 V_{NE} (or 1.1 V_H) point obtained during stability tests required by § 29.175(b). Controllability must be satisfactory for both conditions. If V_{NE} varies with altitude or temperature, V_{NE} for existing ambient conditions is utilized for the test. Extremes of the altitude/temperature envelope should be analyzed and investigated by flight test.

(3) The critical condition for controllability testing in a hover is ordinarily forward CG at maximum weight with minimum power-on rotor RPM. For rearward flight testing of configurations where the forward CG limit varies with weight, low or high gross weight may be critical. Lateral CG limits should also be investigated. A calibrated pace vehicle is needed to assure stabilized flight conditions. Surface winds should be less than 3 knots throughout the test sequence. Testing can be done in higher stabilized wind conditions (gusting less than 3 knots); however, these conditions are very difficult to find and the method is very time consuming due to the necessity of waiting for stabilized winds. Testing in calm winds is preferred. Hover controllability testing should be accomplished with the lowest portion of the rotorcraft at the published hover height above ground level; however, the test altitude above the ground may be increased to provide reasonable ground clearance. Although the necessary yaw response will vary somewhat from model to model, sufficient control power should be available to permit a clearly recognizable yaw response after full directional control displacement when the rotorcraft is held in the most critical position relative to wind.

(4) Prior to engine failure testing, it is mandatory that the pilot be fully aware of his engine, drive system, and rotor limits. These limits were established during previous ground and flight tests and they should be specified in the TIA. Particular attention should be given to minimum stabilized and minimum transient rotor RPM limits. These values must be included in the TIA and should be approached gradually with a build-up in time delay unless the company testing has completely validated all pertinent aspects of engine failure testing. On Category A installations the maximum power output of each engine must be limited so that when an engine fails and the remaining engine(s) assume the additional load, the remaining engine(s) are not damaged by excessive power extraction and over-temping. This is needed for compliance with § 29.903(b). The propulsion engineer should have assured that this feature was properly addressed in the engine and drive system substantiation; however, it must be assumed that for some period of time the pilot may extract maximum available power from the remaining engine(s) when an engine fails during critical flight maneuvers. Substantiation of this feature should be accomplished primarily by engine and drive system ground tests.

(5) Longitudinal cyclic authority at V_{NE} with any power setting must permit suitable nose down pitching of the rotorcraft. If the remaining control travel is considered marginal, tests should include applications up to full control deflection to assess the remaining authority. Some knowledge of the aircraft's response to turbulence is useful in assessing the remaining margin. As a minimum, the rotorcraft must have adequate margin available to overcome a moderate turbulent gust and must not have any divergent characteristic which requires full deflection of the primary recovery control to arrest aircraft motion. If other controls must be utilized to overcome adverse aircraft motion, the results are unacceptable; e.g., if a pitch up tendency resulting from an actual or simulated moderate turbulent gust cannot be satisfactorily overcome by remaining forward cyclic, the use of throttle or collective controls to assist the recovery is not an acceptable procedure; however, the use of lateral cyclic to correct roll in conjunction with forward cyclic to correct pitchup is satisfactory. Obviously during the conduct of these tests, all available techniques should be utilized when the pilot finds himself "out of control." However, compliance with this section requires that recovery must be shown by use of only the primary control for each axis of aircraft motion.

(6) Cyclic control authority in autorotation must be sufficient to allow adequate flare capability and landing under the all engine inoperative requirements of § 29.75(b)(5) and (c). See paragraph AC 29.75.

AC 29.151. <u>§ 29.151 (Amendment 29-24) FLIGHT CONTROLS</u>.

a. <u>Explanation</u>. Excessive breakout or preload in the flight controls produces control system force discontinuities which result in increased workload and even controllability problems for the pilot. Similarly, excessive freeplay results in lost motion which increases pilot workload and, in an extreme case, could lead to a hazardous pilot-induced oscillation. Although in some designs friction can provide a positive contribution to the function of the flight controls (e.g., masking aerodynamic feedback in reversible systems), friction will eventually have a detrimental effect on the pilot's ability to properly control the machine. In the case of an irreversible design equipped with an artificial force feel system in pitch and roll, excessive friction can mask a shallow force gradient making positive stick centering and control force static stability difficult if not impossible to demonstrate. In such an instance, the initial choice of fixes might include implementation of a steeper force gradient or addition of a force preload. Unfortunately, these solutions often lead to the kinds of problems discussed earlier. Care must therefore be exercised during the initial design phase to ensure that the components and characteristics of the flight control system are well matched.

b. Procedures. Regardless of the flight control system sophistication, it is important that the test pilot understand the system configuration prior to flight evaluation. Appropriate mechanical characteristics should be documented. For VFR aircraft, the mechanical characteristics are typically assessed in flight on a qualitative basis. If a controllability or workload problem is identified, a more detailed investigation would be necessary. Since IFR certification rules include specific trim and force requirements, a more quantitative investigation of mechanical characteristics is normally conducted. The constantly varying feedback forces of reversible flight control systems generally make such designs unsuitable for IFR application. Irreversible system mechanical characteristics can often be partially documented on the ground with external hydraulic and electrical power supplies connected to the aircraft. Knowledge of the breakout, friction, and force gradient characteristics prior to flight can be useful to the pilot during flight evaluation of the system.

AC 29.161. § 29.161 TRIM CONTROL.

a. Explanation.

(1) The pilot has many tasks to perform with each hand during sustained flight conditions. The trim requirement is intended to provide the pilot with a reference cyclic control position for the given flight condition, reduce the physical demands to maintain a given flight condition, and allow the pilot to release the cyclic control for brief periods of time to perform other cockpit duties. A primary flight control which can move when released imposes an additional pilot workload by requiring a continuous hands-on condition. It is not intended to require that control forces be reduced to zero by the trim control during dynamic maneuvers such as takeoff acceleration.

(2) A number of devices may be used to produce the necessary trim characteristics. One popular method of meeting this requirement is through the use of control balance springs in conjunction with a small amount of built-in control system friction. Other methods include use of friction, magnetic brakes, bungees, and irreversible mechanical schemes.

(3) This regulation is not intended to require zero friction or zero breakout force in the control system, nor is it intended to require automatic control recentering. The

regulation, in fact, specifically prohibits excessive high friction or high breakout forces which would produce undesirable discontinuities in the primary control force gradient.

b. Procedures.

(1) If comprehensive company flight test data are available, compliance with this requirement can quickly be found by spot checking extreme center of gravity loadings. Trim tests can ordinarily be done during the course of other flight test activities. To conduct the test, simply release the control at the required flight conditions and determine that the control does not move. The words “any appropriate speed” ordinarily include any speed from hover to V_H. If the control system trim device might be subject to temperature or humidity effects, these should be investigated at a minimum of two altitude extremes and during several test phases.

(2) If a pilot controllable variable friction device is incorporated, compliance with this requirement must be shown at the minimum adjustable value. The maximum value of adjustable friction should not completely lock the flight controls.

(3) Continued compliance with this requirement should be assured through a production procedure. If minimum friction or centering springs are used, it is desirable for the manufacturer to include some adjustment capability for production differences. The explanation and procedures discussed here are applicable for VFR approval under § 29.161. For additional IFR trim requirements, refer to AC 29 Appendix B.

AC 29.161A. § 29.161 (Amendment 29-24) TRIM CONTROL.

a. Explanation. Amendment 29-24 to the regulation adds the additional requirement that the trim control be capable of trimming collective forces to zero.

b. Procedures. The trim requirement is intended to allow the pilot to release the controls for brief periods to perform other cockpit duties, and to provide the pilot with a reference cyclic position for the given flight condition. The collective should be balanced so that there is no tendency for the collective pitch to change when the collective is released. Any magnetic clutch, friction brake or similar device which modifies the collective characteristics should be capable of being overpowered by the pilot, when fully applied, without requiring excessive force.

AC 29.171. § 29.171 STABILITY: GENERAL.

a. Explanation. This section is intended to require a manageable pilot workload for the minimum crew under foreseeable operating conditions.

b. Procedures.

(1) Compliance with the requirements of this section can often be obtained for the VFR condition without any specific or designated flight testing. If the rotorcraft is

marginal in regard to pilot strain and fatigue, the FAA/AUTHORITY pilot should be assured, through special tests if necessary, that the aircraft can be satisfactorily flown throughout the maximum endurance capabilities of the rotorcraft including night and turbulence conditions if those are critical. This test should be conducted with minimum required systems in the aircraft and with minimum flight crew.

(2) Reasonable failure conditions which add to pilot workload, strain, and fatigue should be evaluated (electrical, hydraulic and mechanical failures, etc.). The necessary times associated with flight with a failed system must be appropriate to the flight manual procedures for each failure. A failure condition requiring immediate landing would obviously require shorter evaluation time than a condition allowing continued flight to destination.

(3) IFR approvals necessitate a careful evaluation of paragraphs (1) and (2) above. In IFR operations, weather conditions frequently necessitate continued flight to destination or diversion to alternate airports with critical failures. Immediate landing may not be feasible. The evaluating pilot must assure pilot strain and fatigue are acceptable during typical flight profiles for each type of operation to be approved.

AC 29.173. § 29.173 (Amendment 29-24) STATIC LONGITUDINAL STABILITY.

a. Explanation.

(1) This rule contains control system design requirements for both stability and control. Paragraph (a) contains the basic control philosophy necessary for all civil aircraft. Forward motion of the cyclic control must produce increasing speeds and aft motion must result in decreasing speeds. For rotorcraft this is accomplished with throttle and collective held constant. This requirement in no way assures aircraft stability. It is simply a control requirement which speaks to direction of control motion. Rotorcraft with either highly stable or highly unstable static longitudinal stability characteristics can typically comply with the basic requirement for control sense of motion.

(2) The remainder of § 29.173, through reference to § 29.175, contains the basic control position requirements necessary to establish a minimum level of static longitudinal stability. Positive stability is found for conditions of climb, cruise, and autorotation in § 29.175 by requiring a stable stick position gradient through a specified speed range. A defined level of instability is permitted for the hovering condition.

b. Procedures.

(1) The control requirement of this section is so essential to basic flight mechanics that compliance may be found during conventional flight testing for compliance with other portions of the regulations. No special or designated testing should be required.

(2) The procedures necessary to assure compliance with the stability requirements of this section are contained under § 29.175, Demonstration of static longitudinal stability. Refer to paragraph AC 29.175 for an explanation of detailed flight test procedures.

AC 29.175. § 29.175 (Amendment 29-24) DEMONSTRATION OF STATIC LONGITUDINAL STABILITY.

a. Explanation.

(1) This rule incorporates the specific flight requirements for demonstration of static longitudinal stability. Specific loadings, configurations, power levels, and speed ranges are stated for conditions of climb, cruise, autorotation, and hover.

(2) Some rotorcraft in forward flight experience significant changes in engine power with changes in airspeed even though collective and throttle controls are held fixed and altitude remains relatively constant. For these cases, the guidance in § 29.173, which states that throttle and collective pitch must be held constant, is appropriate for administration of this rule, and the specified power in § 29.175(a), (b), and (c) should be considered as power established at initial trim conditions. This will result in slightly higher or lower torque readings at “off trim” conditions. Collective and throttle controls are held constant when obtaining data during climb, cruise, and autorotation tests.

(3) The effects of rotor RPM on autorotative static stability should be determined, and positive stability demonstrated for the most critical RPM. For Category A rotorcraft this requirement may be satisfied at a nominal RPM value. RPM values can be expected to change as airspeed is varied from the “trimmed” condition. Manufacturer’s recommended autorotation airspeed is ordinarily used for trim.

(4) Hovering is considered a flight maneuver for which the pilot repeatedly adjusts collective to maintain an approximately constant altitude above the ground. For hover stability tests, collective and throttle adjustments are made as necessary to maintain an approximately constant height above the ground. Also, a limited amount of negative longitudinal control travel is allowed with changes in speed.

b. Procedures.

(1) Instrumentation.

(i) Sensitive control position instrumentation is mandatory. Engine power parameters should be recorded at trim. For testing of minor modifications or when using a “before and after” method, a tape measure or a stick plotting board may be utilized. A stick plotting board consists of a level surface with a clean sheet of paper on it and attached to the cockpit or seat structure. The installation must not interfere when the flight controls are fully displaced. A recording pencil is attached to the cyclic control by an offsetting arm in such a manner that it can be pushed down on the board to record relative cyclic position at key times during test maneuvers. The figure AC 29.175-1 plot is a typical presentation of longitudinal static stability.

(ii) Other necessary parameters include pitch attitude, pressure altitude, ambient temperature, and indicated airspeed (pace vehicle or theodolite speed for hover tests). For hover tests, hover height (radar altitude if available), and surface winds should be documented. Two-way communications with a pace vehicle is highly desirable. Ground safety equipment is desirable.

(2) <u>Ambient Conditions.</u> Smooth air is necessary for stability testing. Allowable wind conditions for hover stability testing are the same as those for hover controllability tests and are described in that section (paragraph AC 29.151). Extrapolation is covered in paragraph AC 29.53.

(3) <u>Loading.</u> Aft center of gravity (CG) is ordinarily critical for longitudinal stability testing, although high speed flight and hover should be checked at full forward CG and maximum weight. At aft CG, light or heavy weight conditions can be critical. The manufacturer's flight data should be reviewed to determine critical loading conditions.

(4) <u>Conducting The Test.</u>

(i) The rotorcraft should be established in the desired configuration and flight condition (climb, cruise, autorotation) with the required power and rotor speed at the trim airspeed. The collective stick should be fixed in that position, usually by applying sufficient friction to insure that it is not inadvertently moved. For autorotative tests, a rotor speed should be selected so that the variations in rotor speed as airspeed and altitude change do not exceed the allowable limits. This point is recorded as the trim point. Airspeed is then increased or decreased in about 10-knot increments, stabilizing on each speed and recording the data. At least two points on each side of the trim speed should be taken.

(ii) The cruise test should be conducted by varying airspeed around the desired altitude with throttle and collective fixed. This should be accomplished by first determining V_H (level flight speed at maximum continuous power) at the test altitude. Then reduce power to establish a level trimmed condition at 0.9 V_H (or 0.9 V_{NE} if lower). This point is then recorded as the trim point.

(iii) For climb and autorotation tests, conduct fixed collective tests through an altitude band (usually $\pm$2,000 feet), first increasing airspeed as data points are collected, then decreasing speed through the same altitude band. It will probably not be possible to obtain the required data on one pass through the altitude band. If repeated passes are required, a trim point should be taken at the beginning of each pass unless very sensitive collective pitch position information is available in the cockpit. Generally, it will be possible to acquire all the high speed points on one pass and the low speed points on the second.

(iv) If extremely precise results are required, an alternate method of testing can be used to acquire the data at a constant altitude. For cruise, data can be obtained by alternating airspeeds above and below the trim speed to arrive in the vicinity of the test altitude as the point is recorded. This method results in very precise data because collective and throttle are not moved as airspeed is changed at a constant altitude. A typical sequence of speeds that could produce these results would be: 150 (V_H), 135 ($0.9V_H$) trim speed, 125, 145, 115, 155, 105, and 165.

(v) For rotorcraft with high rates of climb, a series of climbs, each at a different speed, may be required through a given altitude, utilizing sensitive instrumentation to assure collective position is the same for each data point. In autorotation, a similar case arises and a series of descents, each at a different speed, may be required through a given altitude band, using sensitive instrumentation to assure a repeatable collective position.

(vi) Hover tests should be conducted by maintaining an approximately constant altitude above the ground at the hover height established for performance purposes. The test altitude above the ground may be increased to provide reasonable ground clearance during rearward flight. Groundspeed is varied using a pace vehicle, theodolite, or other velocity measuring equipment. A pace vehicle is an aid in maintaining an accurate hover height. The pilot can accurately maintain height by controlling his sight picture of the pace vehicle (level with the roof, antenna, etc.). Hover stability tests are ordinarily conducted in conjunction with hover controllability tests because instrumentation and facilities are essentially the same.

(vii) Normally climb, cruise, and autorotation tests should be conducted at low, medium, and high altitudes. See paragraph AC 29.45 for guidance on interpolation and extrapolation. High speed stability has been critical during cold weather testing. In two recent models, V_{NE} at cold temperatures has been limited by the stability requirements of § 29.176(b). Cold weather testing should be accomplished or a conservative approach for advancing blade tip Mach number should be used to limit cold weather V_{NE} to tip Mach number values demonstrated during warm weather testing.

(viii) Hover stability should be verified at low altitude and, if required, at high altitude. Refer to paragraph AC 29.45b(2) for guidance on expansion and extrapolation of altitude.

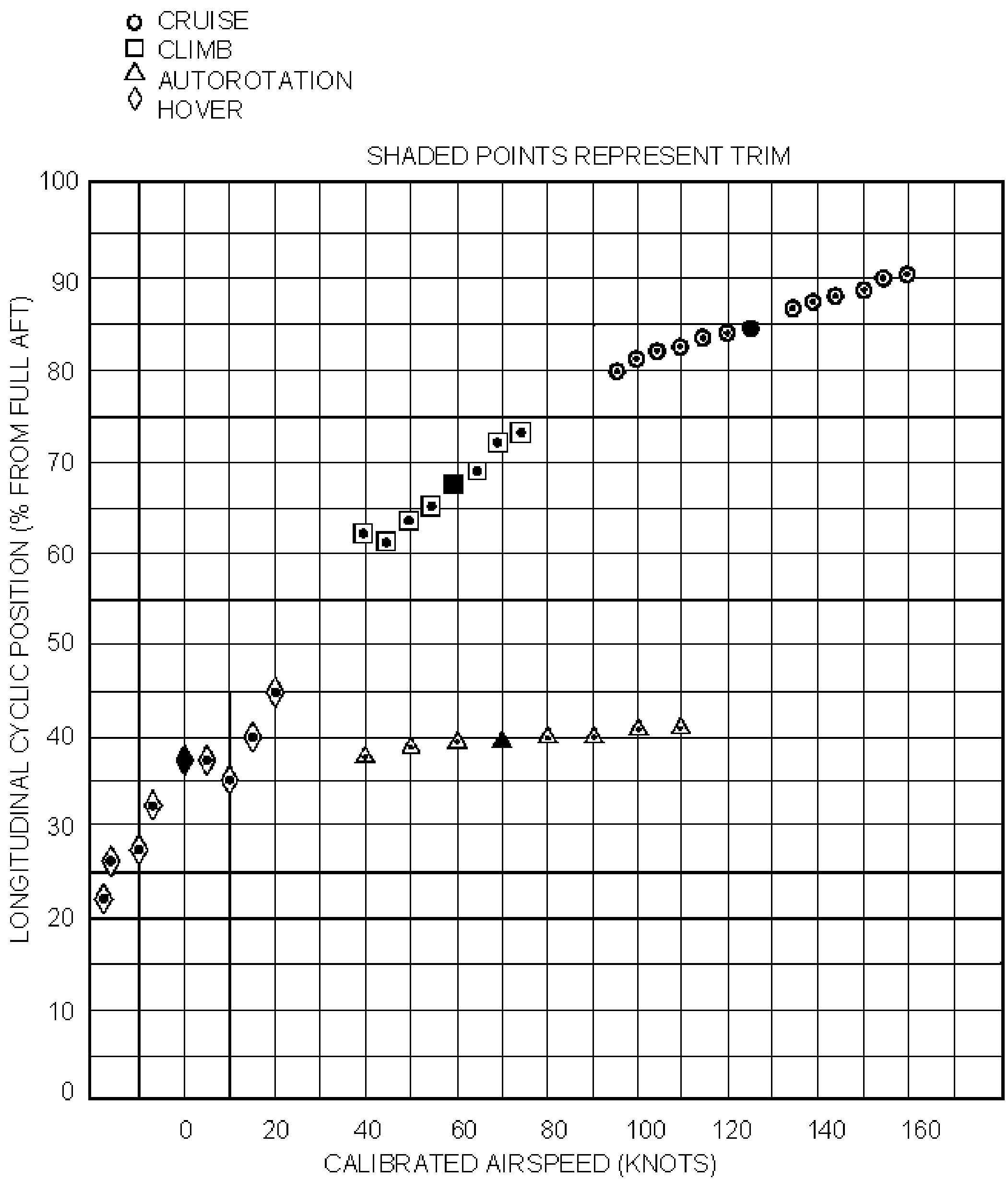


FIGURE AC 29.175-1 STATIC LONGITUDINAL STABILITY

AC 29.177. § 29.177 (Amendment 29-24) STATIC DIRECTIONAL STABILITY.

a. Explanation. This rule requires that positive static directional stability be demonstrated at the trim airspeeds defined in § 29.175. The trim speed for climb is V_Y and for cruise is $0.9V_H$ or $0.9V_{NE}$ (whichever is less). For autorotation that airspeed defined by the midpoint of the speed range specified in § 29.175(c) may be used.

b. Procedures.

(1) Tests for static directional stability require instrumentation for pedal position and sideslip angle. Lateral cyclic control position instrumentation should be provided for IFR certification tests. To obtain accurate sideslip angle and airspeed information, a “yaw boom” is usually installed for the purpose of mounting a sideslip vane and swiveling airspeed pitot head outside the main rotor downwash region of influence. Special care should be taken to ensure that the yaw boom installation has been verified to be structurally adequate and free of dynamic instabilities for all combinations of airspeed and rotor speed likely to be experienced during the static directional evaluation. For some installations, the instrumentation yaw boom may influence the flying qualities of the rotorcraft itself. Thus, it is advisable to correlate yaw string displacement or slip indicator ball widths of skid with yaw boom sideslip angle, and then repeat a few critical points with the yaw boom removed.

(2) For some rotor system designs, the main and tail rotor flapping angle may be a critical instrumentation requirement for static directional testing. Both main and tail rotor flapping may increase dramatically at high airspeeds with increasing sideslip angle. Therefore, for rotor systems exhibiting this characteristic, flapping should be monitored carefully during the sideslip maneuver to avoid exceeding limitations. Static directional stability is normally defined in terms of pedal displacement required to maintain a straight flight path sideslip. A single-rotor rotorcraft flying in coordinated flight will exhibit a small inherent sideslip due to tail rotor thrust and fuselage/main rotor sideforces. This condition is normally taken as trim with the inherent sideslip angle noted. Airspeeds should be the trim values described above. A generally accepted technique follows:

(i) Stabilize at the trim point, and note indicated airspeed.

(ii) Record trim conditions including inherent sideslip. Maintain fixed collective and throttle for the remainder of the maneuver.

(iii) Smoothly yaw the aircraft with directional control and coordinate with lateral control to establish the desired sideslip angle. A steady heading can best be ensured by maintaining a track over a straight landmark on the ground such as a section line or straight segment of powerline or highway.

(iv) Note airspeed immediately upon completion of the yaw maneuver. There may be a small change from the trim airspeed. Fly the new airspeed while maintaining a constant heading, and record indicated airspeed, control positions (directional at a minimum), sideslip angle, rotor speed, rate of descent, amount of ball deflection, and bank angle. The pilot should note the physical sideforce feel experienced. A minimum of two sideslip data points on each side of the trim point should be obtained to adequately define the slope of the pedal displacement versus sideslip angle relationship.

(v) Smoothly return the aircraft to the inherent sideslip angle. Static directional stability plots can be expected to differ slightly on either side of the inherent sideslip angle. Positive static directional stability is indicated by increased left pedal displacement for a larger right sideslip and, conversely, increased right pedal for a larger left sideslip angle.

AC 29.181. § 29.181 (Amendment 29-24) DYNAMIC STABILITY: CATEGORY A ROTORCRAFT.

a. Explanation. This section requires that Transport Category A rotorcraft, certificated under Amendment 24 of FAR 29, demonstrate positive damping for short-period oscillations (5 seconds or less) at forward speeds from V_Y to V_{NE} with the cyclic, collective and directional controls held in the desired test condition or released by the pilot. This requirement would prevent persistent or divergent short-period oscillations and thus alleviate the pilot workload to actively dampen oscillatory motions for all types of operations.

b. Procedures.

(1) Tests for short period dynamic stability are carried out in the same manner as for IFR (reference AC 29 Appendix B) except the oscillation need not be damped as heavily (i.e., to ½ amplitude in not more than one cycle). Similarly pulses and doublets may be used to generate an upset condition that would be expected to be encountered in moderate turbulence for that particular rotorcraft.

(2) Tests should be conducted at the critical gross weight, altitude, center of gravity, rotor RPM, and power conditions during routine climb, cruise, and descent condition for speeds from V_Y to V_{NE}. This test must be conducted with the minimum amount of stability augmentation approved for continued safe flight. Consideration should be given to optional equipment that are to be mounted externally.

(3) This requirement is not applicable to transport category rotorcraft certificated as Category B only. The requirements for this situation are unchanged.

SUBPART B - FLIGHT

GROUND AND WATER HANDLING CHARACTERISTICS

AC 29.231. § 29.231 GENERAL (GROUND AND WATER HANDLING CHARACTERISTICS).

a. Explanation. The rule states: "The rotorcraft must have satisfactory ground and water handling characteristics, including freedom from uncontrollable tendencies in any condition expected in operation." In addition, §§ 29.235, 29.239, and 29.241, contain specific requirements concerning ground and water handling characteristic evaluations.

b. Procedures.

(1) During the flight test program and the F&R program (§ 21.35(b)(2)), the rotorcraft will be subjected to evaluations at various weight and CG conditions. Any uncontrollable tendencies found during these test programs must be corrected.

(2) Controllable or damped vibrations or oscillations on the ground or in the water are acceptable, provided the design limits of the rotorcraft are not exceeded.

(3) Any significant vibration or oscillation characteristics found during tests should be described in the test report, and the rotorcraft flight manual should contain appropriate descriptions and procedures to describe and either avoid or handle significant characteristics.

(4) For rotorcraft equipped with wheel gear, the evaluation should include takeoff, landing, and taxi at the maximum airspeed and ground speed CG extremes. If a nose or tail wheel lock/swivel control is installed, each position should be evaluated for limiting takeoff, landing, and taxi speeds. Maximum substantiated speed values should be included in the RFM as limitations.

(5) For water operations, the wave height and frequency or "sea state" should be included as a limitation or, if no limit was reached during testing, the demonstrated values should be placed in the Performance Section of the Rotorcraft Flight Manual. Information or limits on the allowable "sea state" for rotor startup and shutdown should also be included.

AC 29.235. § 29.235 TAXIING CONDITION.

a. Explanation. The rotorcraft is designed for certain landing load factors (§§ 29.471 and 29.473). The rotorcraft must not attain a load factor in excess of the design load factor when taxied over the roughest ground that may reasonably be expected in normal operation at the expected taxi speeds. This rule applies to wheel landing gear equipped rotorcraft.

b. Procedures. The structural substantiation data contains the allowable design limits for the rotorcraft. A calibrated accelerometer or load factor “g” meter should be installed, as near as practicable to the rotorcraft CG, to record the maximum vertical load factor attained. Instrumentation of the landing gear and/or related structure may also be an acceptable means of showing compliance.

(1) Calibrated instrumentation should be installed to record the maximum loads or maximum vertical load factor attained during the taxi tests.

(2) The taxi surface should be evaluated for compliance with the rule. Corrugated surfaces, as well as broken or uneven surfaces, in accordance with the rule, should be used.

(3) Representative typical taxi speeds, up to the maximum selected by the applicant, should be attained over the selected taxi surfaces.

(4) A light and heavy rotorcraft weight condition should be evaluated.

(5) Limitations appropriate for the rotorcraft design should be included in the flight manual. If these tests indicate that it is unlikely that limit load factors will be attained while taxiing, flight manual limitations may not be necessary.

(6) Pertinent taxi information obtained from these test conditions may be included in normal procedures of the flight manual.

AC 29.239. § 29.239 SPRAY CHARACTERISTICS.

a. Explanation. The intent of this requirement is to evaluate by demonstration that water spray does not obscure visibility (day or night) or damage the rotorcraft during normal waterborne operation (for those rotorcraft which have waterborne or amphibious capability).

b. Procedures.

(1) The following maneuvers should be evaluated in ambient conditions up to the proposed sea state or wave height for operation.

Con-fig.	Condition	Weight	CG	Rotor RPM	Alti-tude	Remarks
1	Taxi	Max	Optional	Max	SL	Speeds up to maximum proposed for water operation.
2	Hover	Max	Opt	Max	-	Determine critical hover height, if any.
3	Takeoff	Max	Opt	Max	SL	Unstick at maximum proposed water operation speed.
4	Land	Max	Opt	Max	SL	Touchdown at maximum proposed for water operation.
5	Shutdown	Opt	Opt	-	SL	Shut down the rotorcraft.
6	Start	Max	Opt	Max	SL	Start engines and release rotor brake.

(2) The maximum sea state or wave height evaluated under this rule should be stated and included in the limitations section of the flight manual.

(3) The effect of saltwater contamination and deterioration of turbine engines and other component parts of the rotorcraft should be considered in accordance with § 29.609 and paragraph AC 29.609. Information on saltwater effect and attendant corrective action should be provided in the flight manual, if appropriate, and in the maintenance manual.

AC 29.241. § 29.241 GROUND RESONANCE.

a. Explanation.

(1) The rule states: “The rotorcraft may have no dangerous tendency to oscillate on the ground with the rotor turning.” This rule is a flight requirement that pertains to demonstrating freedom from dangerous oscillations on the ground. CAR Part 7, predecessor to FAR Part 29, originally contained a “strength requirement,” under § 7.203, requiring ground vibration tests. This test would identify critical vibration frequencies and modes of the rotorcraft. CAR Part 7, Amendment 7-4, effective October 1, 1959, removed this ground vibration requirement because the agency concluded that if any major component has a natural frequency which could be excited by some operating parameter, such a condition would be revealed in the course of other ground and flight tests. The Federal Aviation Administration (FAA) apparently was depending on demonstrations under § 7.131/§ 29.241 and the flight load survey data

(§ 29.571) to satisfy the objective of the vibration test. However, FAR 29, Amendment 29-3, contained new § 29.663 adding reliability and damping action investigation requirements for ground resonance prevention means. A ground vibration survey was not reinstituted by the adoption of § 29.663. Compliance with § 29.663 does require investigation and substantiation as stated. See paragraph AC 29.663.

(2) "Ground resonance" is a mechanical instability of the aircraft while in contact with the ground, often when partially airborne. Stated another way "ground resonance" is a self-excited mechanical instability that involves coupling between the in-plane motion of the rotor blade and the motion of the rotorcraft as a whole on its landing gear (reference "Aerodynamics of the Helicopter," Gessow & Myers, page 308). It is caused by the motion of the blade in the plane of rotation (called in-plane vibration) coupled with a rocking or vertical motion of the aircraft as a whole. The tires, landing gear, and rotor restraint pylon structure act as a spring with a vibration frequency which coincides or couples with the natural in-plane frequency of the blade about a real or effective drag hinge in the plane of rotation. When the frequencies of the two motions (rotor and airframe) approach each other and couple, a violent shaking of the aircraft may occur which, if undamped, could result in the destruction of the rotorcraft.

(3) Ground resonance can occur due to flexibility in the rotor pylon restraint system as well as with landing gear flexibilities. This mode of vibration or resonance can happen in-flight (called air resonance) as well as on the ground and should be addressed in the certification program. The evaluation should include variations in stiffness and damping that could occur in service to the rotor pylon restraints (reference "Ground Vibrations of Helicopters," M.L. Deutsch, JAS, Vol. 13, No. 5, May 1946). See paragraph AC 29.663 for the investigation of the variations.

(4) Ground resonance may be prevented by placing the first order in-plane vibration frequency above the rotor turning speed.

(5) For such configurations which are not susceptible to ground resonance (first order in-plane frequency above rotor turning speed), a simple rotor RPM run-up and run-down with appropriate cyclic control displacement (i.e., excitation of any inherent vibrations) is adequate demonstration that a ground resonance condition does not exist. Unhinged "rigid" rotors, such as Bell Helicopter 2 blade designs, are this type of rotor system.

(6) For configurations that are susceptible to ground resonance (i.e., first in-plane frequency is below the rotor turning speed), ground resonance is generally prevented by dampers on the blade, acting in the plane of rotation, dampers on the landing gear (sometimes serving as oleo struts), or proper placement of the landing gear frequencies combined with rotor and/or landing gear dampers.

(7) Elastomeric components (in the rotor pylon support system, possibly in the landing gear, and possibly in the rotor head) are significantly affected by ambient temperature prior to warm-up. Their damping characteristics require thorough

investigation for the range of rotorcraft operating environment as noted in paragraph AC 29.663.

b. Procedures.

(1) In operation, the resonance characteristics should be checked during takeoff and landing at zero speed and during run-on landings using various power values. Under all conditions, any oscillations which may be introduced should be damped. However, no instability should occur at any operating condition such as during RPM changes from minimum to maximum and idle to maximum. For rotorcraft with wheel gear, uneven taxi surfaces in conjunction with particular taxi speeds, may excite ground resonance and should be evaluated by taxiing on typical surfaces. This evaluation may be conducted in conjunction with tests of § 29.235.

(2) Slow vertical landings for each configuration are made to establish the touchdown collective pitch angle for each rotor speed. For those aircraft equipped with Stability Augmentation Systems (SAS), all ground resonance investigations should be conducted with SAS on and SAS off. This includes the hovering and running takeoffs and landings, taxi tests, and specific ground resonance tests noted herein. Consideration should be given to conducting tests in various SAS configurations such as roll channel on, pitch channel off, where such configurations are possible and authorized.

(3) For each rotorcraft configuration tested, the aircraft should be positioned on the ground in flat pitch with the rotor stabilized at the minimum practical rotational speed, or optionally, at a speed shown analytically to have significant margin from indicated resonant conditions. Control system inputs should be used to disturb the system for evaluation of subsequent damping.

(4) For each incremental increase in rotor speed and for each rotor speed setting at increments of collective pitch settings, cyclic and collective inputs should be investigated prior to proceeding to the next rotor speed setting. These inputs should cover the appropriate range and combinations of amplitude and frequency.

(5) Cyclic pitch inputs should be made, either by the pilot through the cyclic stick, or through a signal generating device working in conjunction with the cyclic controls. For each frequency of input, amplitude of the inputs should be increased incrementally and ultimately should be large enough to generate responses representative of normal ground and flight operation on the rotor and support system. The inputs should continue for a time sufficient to execute five complete counterclockwise circles of the cyclic stick (about neutral) at the selected frequency.

(6) At each amplitude of cyclic input, the excitation frequency should be incrementally increased over the range of the blade in-plane frequency in the fixed system. Rotor speed settings should be increased to 1.05 times the maximum power-on rotor speed. Collective pitch settings should be increased in increments of

not more than 20 percent to maximum collective or alternately to the collective setting required to become partially airborne (when the cyclic is displaced as noted).

(7) Typically, articulated rotor aircraft have natural frequencies on the blade in lag of approximately 0.3 times the power-on main rotor RPM; soft in-plane rotors have natural frequencies approximately 0.7 times the main rotor RPM. Therefore, for example, for a rotorcraft with an in-plane frequency of 0.3/rev, operating at 300 RPM, and with 6 inches of total lateral cyclic stick displacement, the stick should be rotated for 5 revolutions in a 0.6-inch diameter circle at ((1-.03) x 300 RPM) or 3.5 cycles per second to attempt excitation of possible resonant frequencies. At the conclusion of the excitation, the cyclic stick should be returned to the neutral position while continuing the recording of data listed in paragraph b(13).

(8) The complete program should again be repeated with cyclic excitation inputs from the directional and longitudinal controls, if critical for the type of rotorcraft being evaluated.

(9) If onset of ground resonance is encountered, the typical recommended corrective action is to increase the collective pitch and rotor speed and become airborne. However, lowering the collective pitch has been effective for some designs and is considered a satisfactory procedure if resonance can be consistently avoided.

(10) Landings should be made at the maximum touchdown speed proposed with the rotor speed stabilized.

(11) Special Considerations:

(i) The influence of variables including environmental effects, corresponding aircraft component characteristic changes, operational parameters, and surface conditions should be investigated over the ranges proposed for certification. Additionally, the potential of misservicing and possible failure modes should be evaluated. For ground resonance qualification, where practical, variations from the baseline test configuration may be accomplished by either ground run (§ 29.663(b)) requires investigation of probable ranges of damping), analyses, component tests, aircraft shake test, the specification of special operational procedures in the rotorcraft flight manual, or combination thereof. Detailed and rational analyses showing acceptable correlation to the baseline tests, and for which the input parameters were verified by drawings, calculations, component static or dynamic tests, or by aircraft shake tests simulating the conditions/configurations in question, may be used to limit testing to only those variables and operational conditions showing marginal or unacceptable system damping. All operational limitations should be clearly stated in the rotorcraft flight manual. A report of the analytical and/or test results should be permitted per § 29.663.

(ii) Potential instability while airborne, called “air resonance” may occur due to the dynamic coupling of the rotor flexibility and the pylon restraint flexibility. The same considerations apply to air resonance as to ground resonance except that the pylon restraint variables replace the landing gear variables. Air resonance should be addressed in the certification program.

(iii) When operating on the ground, there may be a tendency for the aircraft to exhibit a “ground bounce.” For many configurations, this is a benign, although undesirable phenomenon which may be aggravated by pilot induced oscillations (PIO), particularly if there is little or no friction on the collective.

(12) On rotorcraft with fully articulated rotor heads equipped with landing gear oleos in either skid or wheel configuration, there are tendencies for ground bounce to occur when light on the oleos, either just prior to takeoff or just after landing contact, or during a power assurance check. This bounce may induce ground resonance, particularly if the intensity of the bounce is aggravated by PIO. The corrective action is either to lift off to a hover or to positively lower the collective and remain on the ground.

(13) Instrumentation and Data Acquisition.

(i) Atmospheric Conditions (to be manually noted):

Altitude
OAT
Wind Velocity

(ii) Aircraft Configuration (to be manually noted):

Gross Weight
C.G.
Tire Pressure
Landing Gear Oleo Pressure

(iii) Instrumentation (for recording during test).

Main Rotor RPM.
Time history of cyclic control fore-and-aft and lateral stick position
Time history of collective control stick position
Time history of rotor damper motion*
Time history of pylon component motion*
Time history of landing gear (oleo) motion*
Time history of aircraft motions*

*As required to obtain modal damping

SUBPART B - FLIGHT

MISCELLANEOUS FLIGHT REQUIREMENTS

AC 29.251. § 29.251 VIBRATION.

a. Explanation.

(1) Each part of the rotorcraft must be free from excessive vibration under each appropriate speed and power condition (rule statement).

(2) This flight requirement may be both a qualitative and quantitative flight evaluation. Section 29.571(a) contains the flight load survey requirement that results in accumulation of vibration quantitative data. Section 29.629 generally requires quantitative data to show freedom from flutter for each part of the rotorcraft including control or stabilizing surfaces and rotors. See paragraphs AC 29.571 and 29.629 for these two rules.

(3) Review Case No. 70 (reference FAA Order 8110.6) contains a policy 0statement concerning compliance with this rule. This policy statement is condensed here for convenience:

"The rotorcraft must be capable of attaining a 30° bank angle (turn), at V_{NE}, with maximum continuous power (maximum continuous torque) without encountering excessive roughness/vibration. The FAA/AUTHORITY requires the maneuver demonstration to provide the pilot with some maneuver capability at V_{NE} , and further to provide the pilot some margin away from roughness when operating in turbulence." (This maneuver may result in a descent or a climb.)

(4) Section 29.1505 pertains to V_{NE} determination. Section 29.1509 pertains to rotor speed limits determination. See paragraphs AC 29.1505 and AC 29.1509.

b. Procedures.

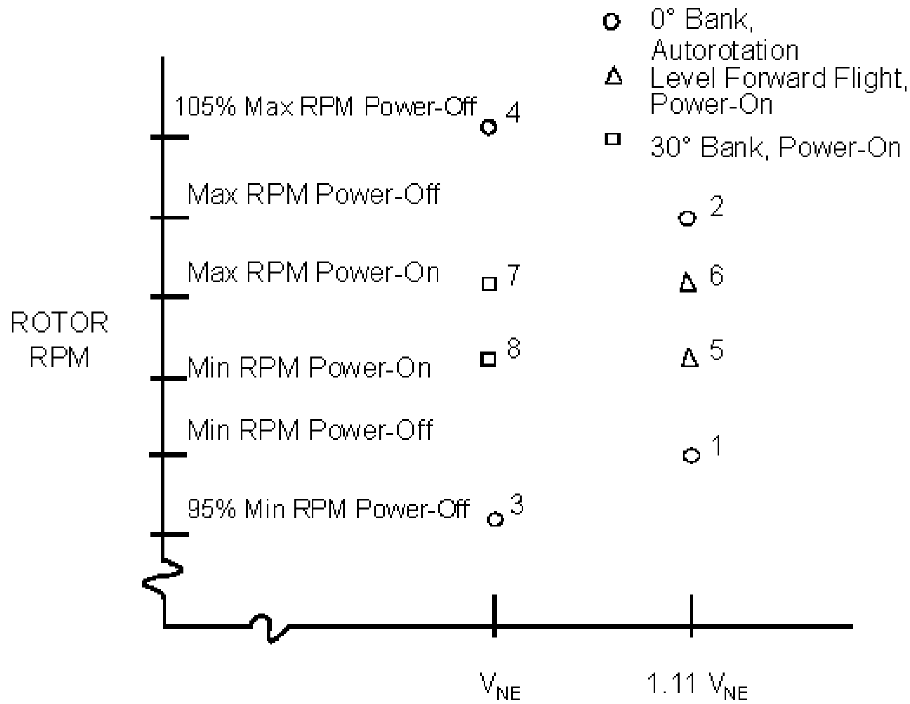
(1) During the company flight test program, the rotorcraft is flown to the appropriate rotor and airspeed limits at several weights to prove that the rotorcraft is free from excessive vibration under appropriate speed, power, and weight conditions. The flight loads survey quantitative data (reference § 29.571) and the applicant's qualitative and quantitative flight test data must also prove compliance with the requirement prior to issuing an authorization for official FAA/AUTHORITY flight tests.

(2) The flight load survey data obtained under § 29.571(a) will contain measured data concerning proof of freedom from flutter and excessive vibration. Pertinent critical flight conditions will be reinvestigated during FAA/AUTHORITY flight tests. The specific condition or conditions necessary to demonstrate compliance with § 29.251 varies with the rotorcraft design, and with the minimum and maximum rotor

speeds, V_{NE} and V_D speeds, and weight and CG position. An illustration of the speed and RPM demonstration is shown in figure AC 29.251-1. Also see subparagraph b(4).

(3) The airspeed and rotor speed limits investigated and established under §§ 29.33, 29.1503, 29.1505, and 29.1509 are also investigated and made a matter of record in the flight loads survey data. During the official FAA/AUTHORITY/TIA flight tests, critical parts of the rotorcraft may have limited instrumentation to reinvestigate and confirm that the critical conditions investigated during the flight load survey are satisfactory and do not result in excessive vibration. Use of instrumentation is optional if the flight loads data (reference paragraph AC 29.571) are conclusive.

(4) FAA policy for certification (Review Case No. 70) requires a “rotor roughness” flight demonstration of a 30° bank angle left and right, at maximum continuous power (MCP) (maximum continuous torque which may be in excess of the maximum continuous temperature limit), at V_{NE}. To provide the pilot with some margin from roughness, the FAA requires maneuver demonstrations of 30°banked turns at V_{NE} without encountering excessive roughness. The maneuver should be conducted with the rotor speed at the minimum RPM and maximum RPM limits. During the flight load survey, this condition should be investigated and data recorded to assure hazardous loads are not encountered for this “unusual” condition. As indicated, the flight condition will be reinvestigated during the FAA/AUTHORITY flight tests. See paragraph b(2) for illustration of this speed and RPM demonstration.



1. Autorotation at 1.11 $V_{NE\ (AR)}$ at minimum placard rotor speed.
2. Autorotation at 1.11 $N_{NE\ (AR)}$ at maximum placard rotor speed.
3. Autorotation at $N_{NE\ (AR)}$ at power-off minimum design limit rotor speed.
4. Autorotation at $N_{NE\ (AR)}$ at power-off maximum design limit rotor speed.
5. Forward flight 1.11 V_{NE} at minimum power-on rotor speed.
6. Forward flight 1.11 V_{NE} at maximum power-on rotor speed.
7. Right and left turn at V_{NE} at maximum power-on rotor speed with 30° bank angle.
8. Right and left turn at V_{NE} at minimum power-on rotor speed with 30° bank angle.
   Note: $V_{NE\ (AR)}$ may be less than V_{NE} .

FIGURE AC 29.251-1 DEMONSTRATION POINTS